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# OH-6A DESIGN AND OPERATIONAL FLIGHT LOADS STUDY

R. Boocock, et al

Summa Corporation

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Army Air Mobility Research and Development Laboratory

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13. ABSTRACT

An analysis and correlation of OH-6A helicopter engineering design values and actual operational values recorded in Southeast Asia is presented. The operational data are based on USAAMRDL Technical Report 71-60, "Flight Loads Investigation of OH-6A Helicopters Operating in Southeast Asia." The parameters involved include mission profiles, rotor drive system, and main and tail rotor fatigue load spectra, damage rates, and service lives. Recommendations are made for additions and changes to improve the structural design criteria for future Army observation helicopters.

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| Mission Profile | | | | | | |
| Spectrum | | | | | | |
| Flight Loads | | | | | | |
| Fatigue Life | | | | | | |
| Design Criteria | | | | | | |
| Light Observation Helicopter | | | | | | |

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This program was conducted under Contract DAAJ02-71-C-0061 with Hughes Helicopter Company.

The information presented herein is the result of an analytical effort to derive improved structural design criteria for observation-type helicopters based upon flight parameters measured on observation helicopters operating in Southeast Asia. This is one of four similar efforts being conducted concurrently to develop improved criteria for utility, crane, and transport as well as observation-type helicopters.

The report has been reviewed by the Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory and is considered to be technically sound. It is published for the exchange of information and the stimulation of future research.

This program was conducted under the technical management of Mr. Herman I. MacDonald, Jr., Technology Applications Division.

# OH-6A DESIGN AND OPERATIONAL
# FLIGHT LOADS STUDY

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# SUMMARY

A study has been conducted by Hughes Helicopters to analyze and correlate OH-6A helicopter engineering design criteria and actual operational flight load data recorded in Southeast Asia. Based on the results of the study, recommendations are made for additions and changes to improve the structural design criteria for future Army observation type helicopters. The work described in this report was performed in five tasks.

In Task I, a mission profile was derived based on OH-6A helicopter operational data recorded in Southeast Asia and contained in USAAMRDL Technical Report 71-60 (Reference 1). Additionally, the mission profile used for design, testing and structural analysis during Hughes' development of the OH-6A helicopter was included, as well as two other current U.S. Army and Navy mission profiles. A comparative analysis noting the differences and deficiencies of the individual profiles was performed. The design mission profile compares most favorably with the operational data.

Main and tail rotor fatigue load data for the design mission profile were obtained from OH-6A flight strain surveys during Task II. Fatigue data were then determined for the operational mission profile defined in the previous task. The two mission profile main rotor fatigue load data show good agreement.

In Task III, main rotor blade and tail rotor blade and the rotor drive system historical changes were analyzed. Design changes found to have been made from the time of original engineering development up to the present are described. The changes are analyzed for their cause-effect relationship with the upgraded OH-6A mission profiles as influenced by increases in gross weight and engine power.

Task IV presents an analysis of parameter peak values. The maximum and minimum one-time occurrences were obtained for selected parameters reported in the operational data. For comparative purpose, similar peak values were obtained from the structural design criteria and from engineering development tests conducted on the OH-6A helicopter. The OH-6A characteristics that influence the magnitude of each of the parameter peak values were evaluated. Additionally, the occurrence of high or peak values has been evaluated for helicopters with limiting characteristics differing from those of the OH-6A.

The results of Tasks I through IV are reviewed in Task V. Indicated revisions to the design criteria for observation type helicopters are presented. Recommendations are made based on the results of each task, and additional general comments and recommendations are presented.

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# INTRODUCTION

This final report is submitted as part of the required documentation pursuant to Contract DAAJ02-72-C-0061, Task IF162204AA8201, between the U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, and Hughes Helicopters (formerly Hughes Tool Company - Aircraft Division). The OH-6A Design and Operational Flight Loads Study presented herein was conducted at the Hughes facility, Culver City, California, from May through December 1972. USAAMRDL Technical Report 71-60, "Flight Loads Investigation of OH-6A Helicopters Operating in Southeast Asia," was used as the basis for the OH-6A operational data. An analysis and correlation between the operational data and OH-6A engineering development values were conducted from the standpoint of mission profiles, main and tail rotor fatigue loads, and damage rates for the purpose of recommending changes or additions to improve Army structural design criteria for future observation type helicopters. The effort was performed in five tasks.

The OH-6A aircraft is basically an all-metal, single-engine, rotary-wing aircraft. It is powered by a T63-A-5A turbine engine driving a four-bladed main rotor and a tail-mounted antitorque rotor through a two-stage, speed reduction transmission. The aircraft is equipped with shock-absorbing landing skids. Primarily an observation aircraft, it is capable of carrying a pilot and three passengers (one of whom may act as a crewmember, copilot, or observer), cargo, or armament subsystem. The aircraft can be equipped with armor for combat operations and can also be used for target acquisition, reconnaissance, and command and control. Dual control provisions allow the aircraft to be flown from either the left- or right-hand pilots compartment seat.

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## TASK I - MISSION PROFILE COMPARISON

### INTRODUCTION

A mission profile or spectrum consists of a complete list of conditions
simulating every type of situation likely to be encountered by a particular
type of helicopter according to the intended usage; for example, utility,
attack, and so forth. The portion of time spent in each condition or per-
centage of occurrence must be specified. In this discussion, three levels
are used for presentation of the mission profile data: (1) mission segments -
this breakdown combines all test conditions into four general categories;
(2) basic conditions - this intermediate level breakdown lists specific, yet
broad, test conditions within each mission segment; and (3) detailed condi-
tions - a further expansion into a detailed condition spectrum. The detailed
conditions comprise all of the actual data points required for a structural
flight test and fatigue analysis, accounting for rotor rpm, density altitude
and other appropriate effects required in establishing the specific test point.
Several mission profiles are presented and compared.

### MISSION PROFILE DETERMINATION

#### Design Mission Profile

The mission profile used for testing and structural analyses during the
design and development of the OH-6A helicopter is described in Reference 2.
Table I presents basic conditions according to mission segment (Refer-
ence 3)* along with the applicable percentage of occurrence. The basic
conditions of this spectrum were obtained from Reference 4. Table II
presents the detailed conditions along with percentage of occurrence for
each. Airspeed, rpm and other pertinent detailed condition information
are also shown.

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*Reference 3 has been used for guidance wherever a particular mission
segment was converted into basic conditions (or vice versa) inasmuch
as a relationship between the two is shown therein.

TABLE I. MISSION PROFILES - BASIC CONDITION

| MISSION SEGMENT Basic Condition | Design (Reference 2) Percentage of Occurrence | Operational (Reference 1) Percentage of Occurrence | AR-56 (Utility) (Reference 6) MISSION SEGMENT Basic Condition | Percentage of Occurrence | AMCP 706-203 (Reference 5) MISSION SEGMENT Basic Condition | Percentage of Occurrence |
|---|---|---|---|---|---|---|
| I. ASCENT | 6.5 | 12.0 | I. ASCENT | 4.5 | I. ASCENT | 6.68 |
| a. Maximum performance takeoff | 0.50 | 0.92 | a. Takeoff | 0.98 | a. Vertical takeoff to 40 ft and accelerate | 0.18 |
|  |  |  |  |  | Normal takeoff and accelerate | 0.37 |
|  |  |  |  |  | Slide takeoff and accelerate | 0.09 |
|  |  |  |  |  | Sling load liftoff | 0.02 |
| b. Climb (takeoff power) | 2.00 | 1.77 | b. Climb (takeoff power) | 0.88 | b. Climb (takeoff power) | 4.36 |
| Climb (maximum continuous power) | 4.0 | 9.31 | Climb (full power) | 2.64 | Climb (maximum continuous power) | 0.04 |
|  |  |  |  |  | Accelerated climb airspeed to cruise | 1.62 |
| II. MANEUVER | 16.6 | 21.0 | II. MANEUVER | 8.08 | II. MANEUVER | 4.84 |
| a. Longitudinal, lateral and pedal reversal, hover | 1.50 | 1.27 | a. Control reversal hover | 0.73 | a. IGE control reversal | 0.07 |
| b. Turn, hover | - | 0.63 | b. Turn, hover | 1.47 | b. IGE turn | 0.26 |
| c. Right Turns - 30, 60, 90% $V_{NE}$ | 3.00 | 3.61 | c. Right turn | 2.20 | c. OGE turn | 2.19 |
| Left Turns - 30, 60, 90% $V_{NE}$ | 3.00 | 3.61 | Left turn | 2.20 | Fire suppression hi-g turn | 0.66 |
| d. Autorotation entry - 30, 60, 90% $V_{NE}$ | 1.50 | 1.89 | d. Autorotation entry | 0.03 | d. Autorotation entry in climb | 0.01 |
|  |  |  |  |  | Autorotation entry in cruise | 0.01 |
| e. Pullup | 1.00 | 1.20 | e. Pullup | 0.18 | e. Cyclic pullup | 0.03 |
|  |  |  |  |  | Fire suppression pullup | 0.86 |
| f. Longitudinal, lateral and pedal reversal | 1.50 | 1.89 | f. Control reversal | 0.59 | f. OGE control reversal | 0.06 |
| g. Simulated power failure | 0.10 | 0.13 | g. - | - | g. - | - |
| h. Pushover | - | 0.64 | h. - | - | h. Fire suppression pushover | - |
| i. Power recovery from autorotation | 0.50 | 0.64 | i. Power recovery from autorotation | 0.03 | i. Power recovery from autorotation approach (IGE) | 0.17 |
|  |  |  |  |  | Power recovery from autorotation minimum power approach, power recovery IGE | 0.01 |
| j. Right turns at 30, 60, 90% $V_{NE}$ autorotation | 1.00 | 1.20 | j. Right turn, autorotation | 0.18 | j. - | 0.50 |
| Left turns at 30, 60, 90% $V_{NE}$ autorotation | 1.00 | 1.20 | Left turn, autorotation | 0.18 |  |  |
| k. Longitudinal, lateral and pedal reversal, autorotation | 1.50 | 1.89 | k. Control reversal autorotation | 0.26 | k. - | - |
| l. Pullup, autorotation | 1.00 | 1.20 | l. Pullup, autorotation | 0.03 | l. - | - |

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TABLE I — Continued

**Design (Reference 2) / Operational (Reference 1)**

| MISSION SEGMENT Basic Condition | Design (Reference 2) Percentage of Occurrence | Operational (Reference 1) Percentage of Occurrence |
|---|---|---|
| III DESCENT | 7.00 | 12.00 |
| a. Partial power descent | 2.00 | 3.43 |
| b. Rapid transition and flare | 3.00 | 2.57 |
| Approach to hover | - | 2.57 |
| c. Autorotation landing, including approach and flare | 2.00 | 3.43 |
| IV STEADY STATE | 69.90 | 55.00 |
| a. Start | 0.25 | 0.23 |
| Shutdown | 0.25 | 0.23 |
| b. Hover, in ground effect (IGE) | 0.50 | 0.43 |
| Hover, out of ground effect (OGE) | - | 0.01 |
| c. Level flight at 20% $V_{NE}$ | 1.00 | 0.92 |
| Level flight at 40% $V_{NE}$ | 3.00 | 1.65 |
| Level flight at 60% $V_{NE}$ | 18.00 | 0.66 |
| Level flight at 80% $V_{NE}$ | 25.30 | 23.10 |
| $V_H$ | 15.00 | 20.96 |
| $V_{NE}$ | 3.00 | 4.19 |
| 111% $V_{NE}$ | 0.60 | 0.16 |
| d. Sideward flight | 0.50 | 0.46 |
| Rearward flight | 0.50 | 0.46 |
| e. Autorotation | 2.00 | 1.52 |

**AR-56 (Utility) Reference 6**

| MISSION SEGMENT Basic Condition | Percentage of Occurrence |
|---|---|
| III DESCENT | 8.21 |
| a. Partial power descent | 4.89 |
| b. Landing approach | 3.06 |
| c. Autorotation landing | 0.26 |
| IV STEADY STATE | 79.21 |
| a. Ground conditions | 0.88 |
| b. Hover | 8.80 |
| c. 20% $V_H$ | 4.40 |
| 40% $V_H$ | 4.40 |
| 50% $V_H$ | 1.76 |
| 60% $V_H$ | 7.04 |
| 70% $V_H$ | 8.80 |
| 80% $V_H$ | 13.20 |
| 90% $V_H$ | 15.85 |
| $V_H$ | 8.80 |
| 115% $V_H$ | 0.88 |
| Power dives | 2.20 |
| d. Sideward flight | 0.88 |
| Rearward flight | 0.44 |
| e. Autorotation | 0.88 |

**AMCP 706-203 (Reference 5)**

| MISSION SEGMENT Basic Condition | Percentage of Occurrence |
|---|---|
| III DESCENT | 7.46 |
| a. Partial power descent | 4.84 |
| b. Slide on landing | 0.11 |
| Approach and landing | 0.46 |
| Decelerate to descent airspeed | 2.02 |
| Sling load landing | 0.02 |
| c. Autorotation approach and landing | 0.01 |
| IV STEADY STATE | 81.00 |
| a. Flat pitch | 2.92 |
| Start | 2.59 |
| Shutdown | 1.29 |
| b. Hover, IGE | 3.24 |
| Hover, OGE | 0.21 |
| c. Loiter airspeed | 18.05 |
| Level at 0.6 $V_{NE}$ | 2.32 |
| Level at 0.7 $V_{NE}$ | 1.92 |
| Cruise at 0.8 $V_{NE}$ | 7.68 |
| Cruise at 0.9 $V_{NE}$ | 21.85 |
| $V_{NE}$ | 17.54 |
| Fire suppression dive | 1.21 |
| d. IGE sideward flight | 0.10 |
| IGE rearward flight | 0.03 |
| e. Autorotation | 0.05 |

## TABLE II. MISSION PROFILES – DETAILED CONDITIONS

| Basic Condition / Detailed Condition | Airspeed | RPM | Remarks | Design Percentage of Occurrence | | Operational Percentage of Occurrence | |
|---|---|---|---|---|---|---|---|
| I  a. Maximum performance takeoff | | | | | | | |
|     1. | 0 | Maximum | | 0.500 | 0.50 | 0.9200 | 0.9200 |
|    b. Climb (takeoff power) | | | | | | | |
|     1. | Best rate of climb | Maximum | | } 2.000 | 2.00 | 0.8850 | 1.7700 |
|     2. | Best rate of climb | Minimum | | | | 0.8850 | |
|    Climb (maximum continuous power) | | | | | | | |
|     3. | Best rate of climb | Maximum | | } 4.000 | 4.00 | 4.6550 | 9.3100 |
|     4. | Best rate of climb | Minimum | | | | 4.6550 | |
| II  a. Longitudinal, lateral and pedal reversal, hover | | | | | | | |
|     1. Longitudinal reversal, hover, rapid | 0 | Maximum | Control Motion ±25% | } 0.150 | 1.50 | 0.0944 | 1.2700 |
|     2. Longitudinal reversal, hover, rapid | 0 | Minimum | ±25% | | | 0.0944 | |
|     3. Longitudinal reversal, hover, slow | 0 | Maximum | ±25% | } 0.350 | | 0.2231 | |
|     4. Longitudinal reversal, hover, slow | 0 | Minimum | ±25% | | | 0.2231 | |
|     5. Lateral reversal, hover, rapid | 0 | Maximum | ±25% | } 0.150 | | 0.0944 | |
|     6. Lateral reversal, hover, rapid | 0 | Minimum | ±25% | | | 0.0944 | |
|     7. Lateral reversal, hover, slow | 0 | Maximum | ±25% | } 0.350 | | 0.2231 | |
|     8. Lateral reversal, hover, slow | 0 | Minimum | ±25% | | | 0.2231 | |
|     9. Pedal reversal, hover, rapid | 0 | Maximum | ±25% | } 0.150 | | - | |
|     10. Pedal reversal, hover, rapid | 0 | Minimum | ±25% | | | - | |
|     11. Pedal reversal, hover, slow | 0 | Maximum | ±25% | } 0.350 | | - | |
|     12. Pedal reversal, hover, slow | 0 | Minimum | ±25% | | | - | |
|    b. Turn, hover | | | | | | | |
|     1. Turn, hover, rapid | 0 | Maximum | Direction Right | - | | | 0.6300 |
|     2. Turn, hover, rapid | 0 | Minimum | Right | - | | 0.0936 | |
|     3. Turn, hover, slow | 0 | Maximum | Right | - | | 0.0936 | |
|     4. Turn, hover, slow | 0 | Minimum | Right | - | | 0.2214 | |
| | | | | | | 0.2214 | |
|    c. Right turn | | | | | | | |
|     1. Buildup g | $0.3\,V_{NE}$ | Maximum | | } 1.000 | 3.00 | 1.0740 | 3.6100 |
|     2. Buildup g | $0.3\,V_{NE}$ | Minimum | | | | 1.0740 | |
|     3. Maximum g | $0.3\,V_{NE}$ | Maximum | | | | 0.0107 | |
|     4. Maximum g | $0.3\,V_{NE}$ | Minimum | | - | | 0.0107 | |
|     5. Buildup g | $0.6\,V_{NE}$ | Maximum | | } 1.000 | | 0.3495 | |
|     6. Buildup g | $0.6\,V_{NE}$ | Minimum | | | | 0.3495 | |
|     7. Maximum g | $0.6\,V_{NE}$ | Maximum | | | | 0.0107 | |
|     8. Maximum g | $0.6\,V_{NE}$ | Minimum | | - | | 0.0107 | |
|     9. Buildup g | $0.9\,V_{NE}$ | Maximum | | - | | 0.3430 | |
|     10. Buildup g | $0.9\,V_{NE}$ | Minimum | | - | | 0.3430 | |
|     11. Buildup g | $0.9\,V_{NE}$ | Maximum | | } 0.965 | | 0.0064 | |
|     12. Buildup g | $0.9\,V_{NE}$ | Minimum | | | | 0.0064 | |
|     13. Maximum g | $0.9\,V_{NE}$ | Maximum | | } 0.035 | | 0.0107 | |
|     14. Maximum g | $0.9\,V_{NE}$ | Minimum | | | | 0.0107 | |
|    Left turn | | | | | | | |
|     15. Buildup g | $0.3\,V_{NE}$ | Maximum | | } 1.000 | 3.00 | 1.0740 | 3.6100 |
|     16. Buildup g | $0.3\,V_{NE}$ | Minimum | | | | 1.0740 | |
|     17. Maximum g | $0.3\,V_{NE}$ | Maximum | | | | 0.0107 | |
|     18. Maximum g | $0.3\,V_{NE}$ | Minimum | | - | | 0.0107 | |
|     19. Buildup g | $0.6\,V_{NE}$ | Maximum | | } 1.000 | | 0.3495 | |
|     20. Buildup g | $0.6\,V_{NE}$ | Minimum | | | | 0.3495 | |
|     21. Maximum g | $0.6\,V_{NE}$ | Maximum | | | | 0.0107 | |
|     22. Maximum g | $0.6\,V_{NE}$ | Minimum | | - | | 0.0107 | |
|     23. Buildup g | $0.9\,V_{NE}$ | Maximum | | - | | 0.3430 | |
|     24. Buildup g | $0.9\,V_{NE}$ | Minimum | | - | | 0.3430 | |
|     25. Buildup g | $0.9\,V_{NE}$ | Maximum | | } 0.965 | | 0.0064 | |
|     26. Buildup g | $0.9\,V_{NE}$ | Minimum | | | | 0.0064 | |
|     27. Maximum g | $0.9\,V_{NE}$ | Maximum | | } 0.035 | | 0.0107 | |
|     28. Maximum g | $0.9\,V_{NE}$ | Minimum | | | | 0.0107 | |
|    d. Autorotation entry | | | | | | | |
|     1. | $0.3\,V_{NE}$ | Maximum | | } 0.100 | 1.50 | 0.3150 | 1.8900 |
|     2. | $0.3\,V_{NE}$ | Minimum | | | | 0.3150 | |
|     3. | $0.6\,V_{NE}$ | Maximum | | } 1.300 | | 0.3150 | |
|     4. | $0.6\,V_{NE}$ | Minimum | | | | 0.3150 | |
|     5. | $0.9\,V_{NE}$ | Maximum | | } 0.100 | | 0.3150 | |
|     6. | $0.9\,V_{NE}$ | Minimum | | | | 0.3150 | |
|    e. Pullup | | | | | | | |
|     1. Buildup g | $0.7\,V_{NE},\ 5\,V_{NE}$ | Maximum | | } 0.500 | 1.00 | 0.4757 | 1.2000 |
|     2. Buildup g | $0.7\,V_{NE},\ 5\,V_{NE}$ | Minimum | | | | 0.4757 | |
|     3. Maximum g | $0.5\,V_{NE}$ | Maximum | | | | 0.0041 | |
|     4. Maximum g | $0.5\,V_{NE}$ | Minimum | | - | | 0.0043 | |
|     5. Buildup g | $0.9\,V_{NE}$ | Maximum | | - | | 0.1114 | |
|     6. Buildup g | $0.9\,V_{NE}$ | Minimum | | | | 0.1114 | |
|     7. Buildup g | $0.9\,V_{NE}$ | Maximum | | } 0.465 | | 0.0043 | |
|     8. Buildup g | $0.9\,V_{NE}$ | Minimum | | | | 0.0041 | |
|     9. Maximum g | $0.9\,V_{NE}$ | Maximum | | } 0.035 | | 0.0043 | |
|     10. Maximum g | $0.9\,V_{NE}$ | Minimum | | | | 0.0043 | |
|    f. Longitudinal, lateral and pedal reversal | | | | | | | |
|     1. Longitudinal reversal, rapid | $V_{NE},\ V_H$ | Maximum | Control Motion ±25% | } 0.150 | 1.50 | 0.0940 | 1.8900 |
|     2. Longitudinal reversal, rapid | $V_{NE},\ V_H$ | Minimum | ±25% | | | 0.0940 | |
|     3. Longitudinal reversal, slow | $V_{NE},\ V_H$ | Maximum | ±25% | } 0.350 | | 0.2210 | |
|     4. Longitudinal reversal, slow | $V_{NE},\ V_H$ | Minimum | ±25% | | | 0.2210 | |
|     5. Lateral reversal, rapid | $V_{NE},\ V_H$ | Maximum | ±25% | } 0.150 | | 0.0940 | |
|     6. Lateral reversal, rapid | $V_{NE},\ V_H$ | Minimum | ±25% | | | 0.0940 | |
|     7. Lateral reversal, slow | $V_{NE},\ V_H$ | Maximum | ±25% | } 0.350 | | 0.2210 | |
|     8. Lateral reversal, slow | $V_{NE},\ V_H$ | Minimum | ±25% | | | 0.2210 | |

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TABLE II — Continued

| Basic Condition Detailed Condition | Airspeed | RPM | Remarks | Design Percentage of Occurrence | Operational Percentage of Occurrence |
|---|---|---|---|---|---|
| II f. 9. Pedal reversal, rapid | $V_{NE}$, $V_H$ | Maximum | ±25% | } 0.1500 | 0.0940 |
| 10. Pedal reversal, rapid | $V_{NE}$, $V_H$ | Minimum | ±25% | | 0.0940 |
| 11. Pedal reversal, slow | $V_{NE}$, $V_H$ | Maximum | ±25% | } 0.3500 | 0.2210 |
| 12. Pedal reversal, slow | $V_{NE}$, $V_H$ | Minimum | ±25% | | 0.2210 |
| g. Simulated Power Failure | | | Density Altitude (Hd) | 0.1000 | 0.1300 |
| 1. | $V_{NE}$ | Minimum | 3,000 ft | 0.0900 | 0.1040 |
| 2. | $V_{NE}$ | Minimum | 7,000 f` | 0.0075 | 0.0200 |
| 3. | $V_{NE}$ | Minimum | 11,000 ft | 0.0025 | 0.0060 |
| h. Pushover | | | | - | 0.6400 |
| 1. Minimum g | 0.3 $V_{NE}$ | Maximum | | - | 0.1900 |
| 2. Minimum g | 0.3 $V_{NE}$ | Minimum | | - | 0.1900 |
| 3. Minimum g | 0.6 $V_{NE}$ | Maximum | | - | 0.0650 |
| 4. Minimum g | 0.6 $V_{NE}$ | Minimum | | - | 0.0650 |
| 5. Minimum g | 0.9 $V_{NE}$ | Maximum | | - | 0.0650 |
| 6. Minimum g | 0.9 $V_{NE}$ | Minimum | | - | 0.0650 |
| i. Power recovery from autorotation | | | | 0.5000 | 0.6400 |
| 1. | 0.5 $V_{NE}$ | Maximum | | } 0.50 | 0.3200 |
| 2. | 0.5 $V_{NE}$ | Minimum | | | 0.3200 |
| j. Right turn, autorotation | | | | 1.0000 | 1.2000 |
| 1. Buildup g | 0.3 $V_{NE}$ | Maximum | | - | 0.6416 |
| 2. Buildup g | 0.3 $V_{NE}$ | Minimum | | | 0.0713 |
| 3. Maximum g | 0.3 $V_{NE}$ | Maximum | | } 0.3333 | 0.0064 |
| 4. Maximum g | 0.3 $V_{NE}$ | Minimum | | | 0.0007 |
| 5. Buildup g | 0.6 $V_{NE}$ | Maximum | | - | 0.2096 |
| 6. Buildup g | 0.6 $V_{NE}$ | Minimum | | - | 0.0233 |
| 7. Maximum g | 0.6 $V_{NE}$ | Maximum | | } 0.3333 | 0.0064 |
| 8. Maximum g | 0.6 $V_{NE}$ | Minimum | | | 0.0007 |
| 9. Buildup g | 0.9 $V_{NE}$ | Maximum | | - | 0.2096 |
| 10. Buildup g | 0.9 $V_{NE}$ | Minimum | | - | 0.0233 |
| 11. Maximum g | 0.9 $V_{NE}$ | Maximum | | } 0.3333 | 0.0064 |
| 12. Maximum g | 0.9 $V_{NE}$ | Minimum | | | 0.0007 |
| Left turn, autorotation | | | | 1.0000 | 1.2000 |
| 13. Buildup g | 0.3 $V_{NE}$ | Maximum | | - | 0.6416 |
| 14. Buildup g | 0.3 $V_{NE}$ | Minimum | | | 0.0713 |
| 15. Maximum g | 0.3 $V_{NE}$ | Maximum | | } 0.3333 | 0.0064 |
| 16. Maximum g | 0.3 $V_{NE}$ | Minimum | | | 0.0007 |
| 17. Buildup g | 0.6 $V_{NE}$ | Maximum | | - | 0.2096 |
| 18. Buildup g | 0.6 $V_{NE}$ | Minimum | | - | 0.0233 |
| 19. Maximum g | 0.6 $V_{NE}$ | Maximum | | } 0.3333 | 0.0064 |
| 20. Maximum g | 0.6 $V_{NE}$ | Minimum | | | 0.0007 |
| 21. Buildup g | 0.9 $V_{NE}$ | Maximum | | - | 0.2096 |
| 22. Buildup g | 0.9 $V_{NE}$ | Minimum | | - | 0.0233 |
| 23. Maximum g | 0.9 $V_{NE}$ | Maximum | | } 0.3333 | 0.0064 |
| 24. Maximum g | 0.9 $V_{NE}$ | Minimum | | | 0.0007 |
| k. Longitudinal, lateral and pedal reversal, autorotation | | | Control Motion | 1.5000 | 1.8900 |
| 1. Longitudinal reversal, rapid | $V_{NE}$, 0.5 $V_{NE}$ | Maximum | ±25% | } 0.1500 | 0.1693 |
| 2. Longitudinal reversal, rapid | $V_{NE}$, 0.5 $V_{NE}$ | Minimum | ±25% | | 0.0188 | 0.0220 |
| 3. Longitudinal reversal, slow | $V_{NE}$, 0.5 $V_{NE}$ | Maximum | ±25% | } 0.3500 | 0.4002 |
| 4. Longitudinal reversal, slow | $V_{NE}$, 0.5 $V_{NE}$ | Minimum | ±25% | | 0.0445 |
| 5. Lateral reversal, rapid | $V_{NE}$, 0.5 $V_{NE}$ | Maximum | ±25% | } 0.1500 | 0.1693 |
| 6. Lateral reversal, rapid | $V_{NE}$, 0.5 $V_{NE}$ | Minimum | ±25% | | 0.0188 |
| 7. Lateral reversal, slow | $V_{NE}$, 0.5 $V_{NE}$ | Maximum | ±25% | } 0.3500 | 0.4002 |
| 8. Lateral reversal, slow | $V_{NE}$, 0.5 $V_{NE}$ | Minimum | ±25% | | 0.0445 |
| 9. Pedal reversal, rapid | 0.5 $V_{NE}$ | Maximum | ±25% | | 0.1685 |
| 10. Pedal reversal, rapid | $V_{NE}$ | Maximum | ±25% | 0.1000 | - |
| 11. Pedal reversal, rapid | 0.5 $V_{NE}$ | Minimum | ±25% | 0.0010, | 0.0188 |
| 12. Pedal reversal, rapid | $V_{NE}$ | Minimum | ±25% | | |
| 13. Pedal reversal, slow | 0.5 $V_{NE}$ | Maximum | ±25% | | 0.3934 |
| 14. Pedal reversal, slow | $V_{NE}$ | Maximum | ±25% | } 0.3950, | - |
| 15. Pedal reversal, slow | 0.5 $V_{NE}$ | Minimum | ±25% | 0.0040 | 0.0436 |
| 16. Pedal reversal, slow | $V_{NE}$ | Minimum | ±25% | | - |
| l. Pullup, autorotation | | | | 1.0000 | 1.2000 |
| 1. Buildup g | 0.5 $V_{NE}$ | Maximum | | - | 0.8563 |
| 2. Buildup g | 0.5 $V_{NE}$ | Minimum | | - | 0.0951 |
| 3. Maximum g | 0.5 $V_{NE}$ | Maximum | | - | 0.0077 |
| 4. Maximum g | 0.5 $V_{NE}$ | Minimum | | - | 0.0009 |
| 5. Buildup g | 0.9 $V_{NE}$ | Maximum | | } 0.9000 | 0.2083 |
| 6. Buildup g | 0.9 $V_{NE}$ | Minimum | | | 0.0231 |
| 7. Maximum g | 0.9 $V_{NE}$ | Maximum | | } 0.1000 | 0.0077 |
| 8. Maximum g | 0.9 $V_{NE}$ | Minimum | | | 0.0009 |
| III a. Partial power descent | | | | 2.0000 | 3.4300 |
| 1. | 30 knots to 0 | Maximum | | } 2.0000 | - |
| 2. | 30 knots to 0 | Minimum | | | - |
| 3. includes vertical descent, | 30 knots | Maximum | | - | 1.5450 |
| 4. zero forward speed | 30 knots | Minimum | | - | 1.5450 |
| 5. (vortex ring state) | 0 | Maximum | | - | 0.1700 |
| 6. | 0 | Minimum | | - | 0.1700 |
| b. Rapid transition and flare | | | | 3.0000 | 2.5700 |
| 1. | 0 to 0.5 $V_{NE}$ to 0 | Maximum | | 2.0000 | 1.7100 |
| 2. | 0 to 0.5 $V_{NE}$ to 0 | Minimum | | 1.0000 | 0.8600 |
| Approach to hover | | | | - | 2.5700 |
| 3. | 0 | Maximum | | - | 0.8550 |
| 4. | 0 | Minimum | | - | 0.4300 |
| 5. Different pilot than 3 and 4 | 0 | Maximum | | - | 0.8550 |
| 6. Different pilot than 3 and 4 | 0 | Minimum | | - | 0.4300 |

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# TABLE II – Continued

| Basic Condition / Detailed Condition | Airspeed | RPM | Remarks | Design Percentage of Occurrence | Operational Percentage of Occurrence |
|---|---|---|---|---|---|
| **III c. Autorotation landing** | | | | 2.0000 | 3.4300 |
| 1. | 0 | - | | 2.0000 | 3.4300 |
| **IV a. Start** | | | | 0.2500 | 0.2300 |
| 1. Engine start and rpm sweep | 0 | 0 to Maximum | | 0.2500 | 0.2300 |
| Shutdown | | | | 0.2500 | 0.2300 |
| 2. Shutdown to rotor stopped | 0 | Maximum to 0 | | 0.2500 | 0.2300 |
| **b. Hover, IGE** | | | | 0.5000 | 0.4300 |
| 1. IGE | 0 | Maximum | | } 0.5000 | 0.2150 |
| 2. IGE | 0 | Minimum | | | 0.2150 |
| Hover, OGE | | | | - | 0.0300 |
| 3. OGE | 0 | Maximum | | - | 0.0150 |
| 4. OGE | 0 | Minimum | | - | 0.0150 |
| **c. Level flight ~ 20% $V_{NE}$** | | | | 1.0000 | 0.9200 |
| 1. | $0.2\ V_{NE}$ | Maximum | 3,000 ft $H_d$ | } 0.8000 | 0.3680 |
| 2. | $0.2\ V_{NE}$ | Minimum | 3,000 ft $H_d$ | | 0.3680 |
| 3. | $0.2\ V_{NE}$ | Maximum | 7,000 ft $H_d$ | } 0.1500 | 0.0690 |
| 4. | $0.2\ V_{NE}$ | Minimum | 7,000 ft $H_d$ | | 0.0690 |
| 5. | $0.2\ V_{NE}$ | Maximum | 11,000 ft $H_d$ | } 0.0500 | 0.0230 |
| 6. | $0.2\ V_{NE}$ | Minimum | 11,000 ft $H_d$ | | 0.0230 |
| Level Flight ~ 40% $V_{NE}$ | | | | 3.0000 | 1.6500 |
| 7. | $0.4\ V_{NE}$ | Maximum | 3,000 ft $H_d$ | } 2.4000 | 0.6600 |
| 8. | $0.4\ V_{NE}$ | Minimum | 3,000 ft $H_d$ | | 0.6600 |
| 9. | $0.4\ V_{NE}$ | Maximum | 7,000 ft $H_d$ | } 0.4500 | 0.1218 |
| 10. | $0.4\ V_{NE}$ | Minimum | 7,000 ft $H_d$ | | 0.1217 |
| 11. | $0.4\ V_{NE}$ | Maximum | 11,000 ft $H_d$ | } 0.1500 | 0.0413 |
| 12. | $0.4\ V_{NE}$ | Minimum | 11,000 ft $H_d$ | | 0.0412 |
| Level Flight ~ 60% $V_{NE}$ | | | | 18.0000 | 0.6600 |
| 13. 1.15 g pullup | $0.6\ V_{NE}$ | Maximum | 3,000 ft $H_d$ | } 14.4000 | 0.2640 |
| 14. 1.15 g pullup | $0.6\ V_{NE}$ | Minimum | 3,000 ft $H_d$ | | 0.2640 |
| 15. 1.15 g pullup | $0.6\ V_{NE}$ | Maximum | 7,000 ft $H_d$ | } 2.7000 | 0.0495 |
| 16. 1.15 g pullup | $0.6\ V_{NE}$ | Minimum | 7,000 ft $H_d$ | | 0.0495 |
| 17. 1.15 g pullup | $0.6\ V_{NE}$ | Maximum | 11,000 ft $H_d$ | } 0.9000 | 0.0165 |
| 18. 1.15 g pullup | $0.6\ V_{NE}$ | Minimum | 11,000 ft $H_d$ | | 0.0165 |
| Level Flight ~ 80% $V_{NE}$ | | | | 25.3000 | 24.1000 |
| 19. 1.15 g pullup, 1.30 g pullup | $0.8\ V_{NE}$ | Maximum | 3,000 ft $H_d$ | } 20.3000 | 9.2400 |
| 20. 1.15 g pullup, 1.30 g pullup | $0.8\ V_{NE}$ | Minimum | 3,000 ft $H_d$ | | 9.2400 |
| 21. 1.15 g pullup | $0.8\ V_{NE}$ | Maximum | 7,000 ft $H_d$ | } 3.7500 | 1.7326 |
| 22. 1.15 g pullup | $0.8\ V_{NE}$ | Minimum | 7,000 ft $H_d$ | | 1.7324 |
| 23. 1.15 g pullup | $0.8\ V_{NE}$ | Maximum | 11,000 ft $H_d$ | } 1.2500 | 0.5776 |
| 24. 1.15 g pullup | $0.8\ V_{NE}$ | Minimum | 11,000 ft $H_d$ | | 0.5774 |
| Level Flight ~ $V_H$ | | | | 15.0000 | 20.9600 |
| 25. 1.10 g pullup | $V_H$ | Maximum | 3,000 ft $H_d$ | } 12.0000 | 8.3840 |
| 26. 1.10 g pullup | $V_H$ | Minimum | 3,000 ft $H_d$ | | 8.3840 |
| 27. 1.10 g pullup | $V_H$ | Maximum | 7,000 ft $H_d$ | } 2.2500 | 1.5721 |
| 28. 1.10 g pullup | $V_H$ | Minimum | 7,000 ft $H_d$ | | 1.5719 |
| 29. 1.10 g pullup | $V_H$ | Maximum | 11,000 ft $H_d$ | } 0.7500 | 0.5241 |
| 30. 1.10 g pullup | $V_H$ | Minimum | 11,000 ft $H_d$ | | 0.5239 |
| $V_{NE}$ | | | | 3.0000 | 4.1900 |
| 31. 1.10 g pullup | $V_{NE}$ | Maximum | 3,000 ft $H_d$ | } 2.4000 | 1.6760 |
| 32. 1.10 g pullup | $V_{NE}$ | Minimum | 3,000 ft $H_d$ | | 1.6760 |
| 33. 1.10 g pullup | $V_{NE}$ | Maximum | 7,000 ft $H_d$ | } 0.4500 | 0.3142 |
| 34. 1.10 g pullup | $V_{NE}$ | Minimum | 7,000 ft $H_d$ | | 0.3142 |
| 35. 1.10 g pullup | $V_{NE}$ | Maximum | 11,000 ft $H_d$ | } 0.1500 | 0.1048 |
| 36. 1.10 g pullup | $V_{NE}$ | Minimum | 11,000 ft $H_d$ | | 0.1048 |
| 111% $V_{NE}$ | | | | 0.6000 | 0.1600 |
| 37. | 111% $V_{NE}$ | Maximum | 3,000 ft $H_d$ | 0.3200 | 0.0640 |
| 38. | 111% $V_{NE}$ | Minimum | 3,000 ft $H_d$ | 0.1800 | 0.0640 |
| 39. | 111% $V_{NE}$ | Maximum | 7,000 ft $H_d$ | 0.0600 | 0.0121 |
| 40. | 111% $V_{NE}$ | Minimum | 7,000 ft $H_d$ | 0.0150 | 0.0119 |
| 41. | 111% $V_{NE}$ | Maximum | 11,000 ft $H_d$ | 0.0200 | 0.0041 |
| 42. | 111% $V_{NE}$ | Minimum | 11,000 ft $H_d$ | 0.0050 | 0.0039 |
| **d. Sideward Flight** | | | | 0.5000 | 0.4600 |
| 1. Right sideward | 0 to 35 knots | Maximum | | } | - |
| 2. Right sideward | 0 to 35 knots | Minimum | | 0.5000 | - |
| 3. Left sideward | 0 to 35 knots | Maximum | | | - |
| 4. Left sideward | 0 to 35 knots | Minimum | | | |
| 5. Right sideward | 5 mph | Maximum | | - | 0.0578 |
| 6. Right sideward | 5 mph | Minimum | | - | 0.0568 |
| 7. Right sideward | 10 mph | Maximum | | - | 0.0278 |
| 8. Right sideward | 10 mph | Minimum | | - | 0.0278 |
| 9. Right sideward | 15 mph | Maximum | | - | 0.0160 |
| 10. Right sideward | 15 mph | Minimum | | - | 0.0160 |
| 11. Right sideward | 20 mph | Maximum | | - | 0.0096 |
| 12. Right sideward | 20 mph | Minimum | | - | 0.0096 |
| 13. Right sideward | 25 mph | Maximum | | - | 0.0043 |
| 14. Right sideward | 25 mph | Minimum | | - | 0.0043 |
| 15. Left sideward | 5 mph | Maximum | | - | 0.0578 |
| 16. Left sideward | 5 mph | Minimum | | - | 0.0568 |
| 17. Left sideward | 10 mph | Maximum | | - | 0.0278 |
| 18. Left sideward | 10 mph | Minimum | | - | 0.0278 |
| 19. Left sideward | 15 mph | Maximum | | - | 0.0160 |
| 20. Left sideward | 15 mph | Minimum | | - | 0.0160 |
| 21. Left sideward | 20 mph | Maximum | | - | 0.0096 |
| 22. Left sideward | 20 mph | Minimum | | - | 0.0096 |
| 23. Left sideward | 25 mph | Maximum | | - | 0.0043 |
| 24. Left sideward | 25 mph | Minimum | | - | 0.0043 |

TABLE II — Continued

| Basic Condition Detailed Condition | Airspeed | RPM | Remarks | Design Percentage of Occurrence | Operational Percentage of Occurrence |
|---|---|---|---|---|---|
| IV d. Rearward flight | | | | 0.5000 | 0.4600 |
| 25. Rearward | 0 to 35 knots | Maximum | | } 0.50 | - |
| 26. Rearward | 0 to 35 knots | Minimum | | | - |
| 27. Rearward | 5 mph | Maximum | | - | 0.1145 |
| 28. Rearward | 5 mph | Minimum | | - | 0.1145 |
| 29. Rearward | 10 mph | Maximum | | - | 0.0556 |
| 30. Rearward | 10 mph | Minimum | | - | 0.055 . |
| 31. Rearward | 15 mph | Maximum | | - | 0.0421 |
| 32. Rearward | 15 mph | Minimum | | - | 0.0421 |
| 33. Rearward | 20 mph | Maximum | | - | 0.0182 |
| 34. Rearward | 20 mph | Minimum | | - | 0.0182 |
| 35. Rearward | 25 mph | Maximum | | - | 0.0096 |
| 36. Rearward | 25 mph | Minimum | | - | 0.0096 |
| e. Autorotation | | | | 2.0000 | 1.5200 |
| 1. | $0.5 V_{NE}$, $0.6 V_{NE}$ | Maximum | 3,000 ft $H_d$ | } 1.8000 | 1.2312 |
| 2. | $0.5 V_{NE}$, $0.6 V_{NE}$ | Minimum | 3,000 ft $H_d$ | | 0.1368 |
| 3. | $111\% V_{NE}$ | Maximum | 3,000 ft $H_d$ | 0 | 0 |
| 4. | $V_{NE}$ | 105% Maximum | 3,000 ft $H_d$ | 0 | 0 |
| 5. | $111\% V_{NE}$ | Minimum | 3,000 ft $H_d$ | 0 | 0 |
| 6. | $V_{NE}$ | Minimum | 3,000 ft $H_d$ | 0.0800 | 0.0608 |
| 7. | $V_{NE}$ | 95% Minimum | 3,000 ft $H_d$ | 0 | 0 |
| 8. | $V_{NE}$ | Maximum | 3,000 ft $H_d$ | 0.0800 | 0.0608 |
| 9. | $111\% V_{NE}$ | Maximum | 7,000 ft $H_d$ | 0 | 0 |
| 10. | $V_{NE}$ | 105% Maximum | 7,000 ft $H_d$ | 0 | 0 |
| 11. | $111\% V_{NE}$ | Minimum | 7,000 ft $H_d$ | 0 | 0 |
| 12. | $V_{NE}$ | 95% Minimum | 7,000 ft $H_d$ | 0 | 0 |
| 13. | $V_{NE}$ | Maximum | 7,000 ft $H_d$ | 0.0150 | 0.0114 |
| 14. | $111\% V_{NE}$ | Maximum | 11,000 ft $H_d$ | 0 | 0 |
| 15. | $V_{NE}$ | 105% Maximum | 11,000 ft $H_d$ | 0 | 0 |
| 16. | $111\% V_{NE}$ | Minimum | 11,000 ft $H_d$ | 0 | 0 |
| 17. | $V_{NE}$ | 95% Minimum | 11,000 ft $H_d$ | 0 | 0 |
| 18. | $V_{NE}$ | Maximum | 11,000 ft $H_d$ | 0.0050 | 0.0039 |
| 19. Gross weight = minimum with instrumentation | $V_{NE}$ | Minimum | 11,000 ft $H_d$ | 0.0150 | 0.0113 |
| 20. Gross weight = midway between above and maximum | $V_{NE}$ | Minimum | 7,000 ft $H_d$ | 0.0050 | 0.0038 |

NOTES: (1) Braces { } indicate that the percentage of occurrence shown is the total for the conditions so marked.

## Operational Mission Profile

The initial step in deriving a spectrum based on the OH-6A operational data would ordinarily be one of distributing the time of the four mission segments (discussed in Reference 1) into basic conditions. However, prior to developing a breakdown of each mission segment into basic conditions, a preliminary evaluation of Reference 1 revealed a questionable statistic; that is, percentage of occurrence in the maneuver mission segment was 51 percent. In view of previous Hughes experience, the figure appeared substantially greater than expected. The mission segment definitions presented in Reference 2 are quite vague. Furthermore, the method used to determine the mission segment to which each specific portion of test data was assigned, is not adequately explained.

Consequently, an analysis was conducted to explain the large percentage of time spent in the maneuver mission segment. Figure 12.c of Reference 1 shows that 0.02 hour was required to reach or exceed 1.3g. The number of occurrences that reach or exceed 1.3g is the reciprocal of this time and is, therefore, known for any specified time period; for example, 50,000 occurrences in 1000 hours of maneuver mission segment time.

The basic conditions that contribute load factors in excess of 1.3g are turns, pullups and longitudinal reversals in forward flight. Based on an evaluation of typical flight conditions by Hughes test pilots during several programs, the duration of these maneuvers is assumed to be 6, 3 and 3 seconds, respectively. According to design data, turns account for 73 percent of the occurrences, pullups and reversals the remaining 27 percent.** Using this information, calculations were made to determine the total time that might reasonably be expected to be spent in the aforementioned load factor-producing conditions in 1000 hours of mission profile time. This calculated percentage of occurrence is 3.7 percent and covers all load factor occurrences equal to or greater than 1.3g in the operational maneuver mission segment. The design mission profile data presented in Table I indicate that 66 percent of the time in the maneuver mission segment is spent in turns, pullups and forward flight longitudinal reversals. Applying this factor to the 51 percent time indicated for the total operational maneuver mission segment means that 33.7 percent of the time is used for turns, pullups and longitudinal reversals according to the definitions stated in Reference 1. Of this, 3.7 percent (as calculated above) is at 1.3g or greater. Therefore, the remaining 30 percent of the time (33.7 percent - 3.7 percent) must be spent in mild turns, pullups and reversal maneuvers where the load factor is less than 1.3g. Mild maneuvers of this type have been covered as part of the steady-state mission segment conditions in previous Hughes spectrum analyses. Consequently, the maneuver mission segment percentage of occurrence has been reduced by 30 percent; that is, from 51 percent to 21 percent, and the steady-state segment increased by 30 percent, from 25 to 55 percent. The adjusted operational percentage of occurrence values for all four mission segments are shown in Table I.

Basic conditions for the operational spectrum are listed under the appropriate mission segment in Table I. The conditions are identical to the design basic conditions with minor exceptions as noted. The mission segment percent time has been distributed among the appropriate basic conditions.

For the operational ascent mission segment, the basic condition percentage of occurrence were determined by ratioing the design basic condition values by the operational and design spectrum ascent segment totals; that

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**From Table I (design profile), turns percentage of occurrence = 3 + 3 + 1 + 1 = 8 or 73 percent of 11 percent total, and pullups and longitudinal reversal (one-third of reversals) percentage of occurrence = 1 + 0.5 + 0.5 + 1 = 3 or 27 percent of 11 percent total.

is, 12/6.5. However, the time split for the takeoff and maximum continuous power climb conditions was further adjusted based on Figure 10.a of Reference 1, which shows engine torque versus percentage of time for the ascent segment.

To obtain an operational spectrum percentage of occurrence for the basic conditions shown under the maneuver mission segment in Table I, the basic condition percentage of occurrence values from the design column were ratioed by the operational and design maneuver mission segment figures, 21.0/16.5. The design values prior to modification were as shown in Table I, except as discussed below.

Hover turns were assigned the 0.5 percent previously applied to hover pedal reversals. As was determined during OH-6A developmental tests, the hover turn which is, in fact, a slow pedal reversal, is considered to be a more critical and realistic test condition. Since the pushover maneuver was not accounted for in the original OH-6A development spectrum, a percentage of occurrence for pushovers was determined by using Figure 12.c of Reference 1. Using an analysis technique similar to that previously discussed for turns, pullups and reversals, and covering all pushovers with load factors less than 0.8g, a 0.5 percentage of occurrence was calculated. The design values for turns and pullups were proportionally reduced in order to provide the time allotted to pushovers.

The design percentage of occurrence figures were ratioed by 12/7 (technique similar to ascent segment) to obtain the operational descent mission segment basic condition values. The design mission profile rapid transition and flare basic condition was expanded to include an approach to hover. The 3-percent occurrence noted in the design column was split between the two conditions before the calculations were performed, based on previous Hughes experience.

The curve of cumulative percentage of time versus airspeed was created from the steady-state mission segment data of Reference 1 (refer to Figure 1). Specific airspeed ranges defined by the basic conditions were identified, and a percentage of occurrence was assigned to each condition based on the time spent in that speed range. Based on the wide range of gross weights, rpm's and altitudes shown in Reference 1, the maximum level flight airspeed ($V_H$) will be considered equal to the never exceed airspeed ($V_{NE}$). Design spectrum basic condition ratios were employed, to determine the autorotation and zero airspeed (ground, hover, sideward and rearward) condition percentage of occurrences. The hover out
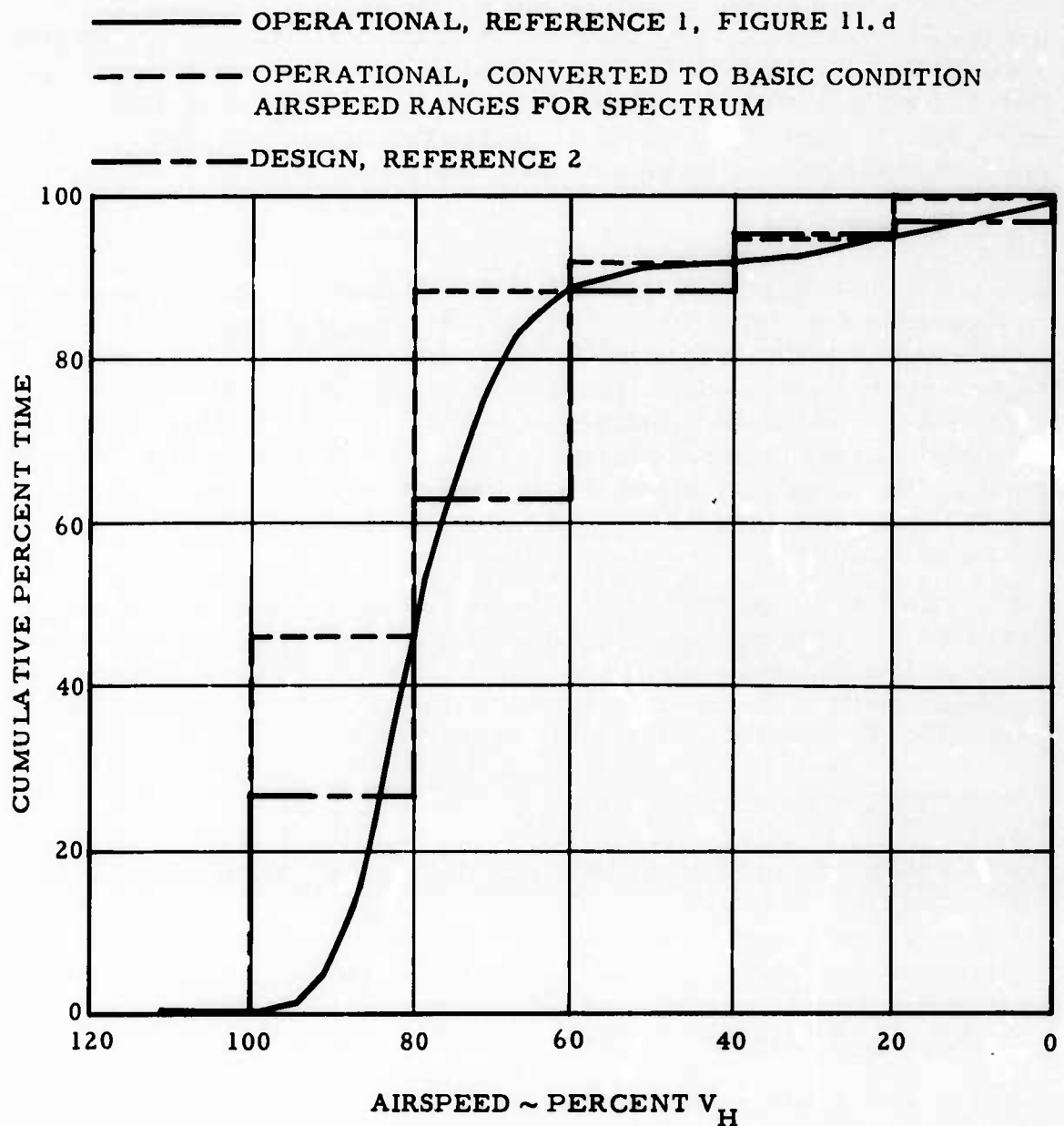
Figure 1. Cumulative Airspeed Data - Steady-State Mission Segment.

of ground effect (OGE) basic condition was added to the original design spectrum. The ratio used for hover in ground effect (IGE) to OGE was taken from Reference 5.

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A listing of the detailed conditions for the operational mission profile is presented in Table II. The design mission profile data, subsequent Hughes spectrum analysis and Reference 1 were used as aids in defining the detailed conditions and percentage of occurrences for the operational mission profile. Table III contains an explanation of the operational spectrum detailed conditions percentage of occurrence distribution.

## AR-56 Mission Profile

The utility helicopter spectrum was selected from the several available in Reference 6 as being the most appropriate for this presentation. The basic conditions shown in the reference material are divided between maneuvers per one hundred flight hours and percentage of service life conditions. The AR-56 spectrum data shown on Table I were obtained by converting the former to time required, then combining them with the time for the percentage of service life conditions, and finally ratioing all of the combined conditions to 100 percent occurrence. The procedure is shown in Table IV.

Detailed conditions are not shown for this mission profile or the one described in the following paragraph inasmuch as the breakdown from basic to detailed conditions would be the same as that for the operational mission profile. The basic condition percentage of occurrences would be proportionally distributed the same as in the operational breakdown.

## AMCP 706-203 Mission Profile

Table I shows the AMCP 706-203 spectrum mission segments and basic conditions from Reference 5 in the final section. The conditions descriptions have been reworded in order to make them compatible with the single engine helicopter data presented in the rest of the table.

## MISSION PROFILE COMPARISON

Tables I and II, which present the mission profile data from the four sources discussed earlier, were organized in a manner to enable direct comparisons of the spectrums. In Table I, the basic conditions within each mission segment have been alphabetized in related sets. The mission segment and basic condition comparisons that follow refer to Table I. The detailed condition comparison is based on Table II.

### Mission Segment Breakdown

Comparing the percentage of occurrence values of the mission segment headings, the design spectrum most nearly matches the operational data,

# TABLE III. DETAILED OPERATIONAL SPECTRUM CONDITIONS PERCENTAGE OF OCCURRENCE DISTRIBUTION

| Table II Item Numbers | Parameter | Percentage of Occurrence Distribution | Explanation |
|---|---|---|---|
| As applicable (except I-a and III-b) | Power on rpm | Maximum ~ 50% Minimum ~ 50% | Based on design spectrum. Reference 1 Figure 6 rpm versus time shows poor selection of rpm "bins", also 3 $\sigma$ variation (accuracy) is ±6 rpm which is greater than ±1%. |
| All (except II-g, IV-c) | Altitude | Sea level - 3000 ft $H_d$ ~ 100% | Based on design da a which essentially matches Reference 1, Figure 7 shows that over 90% of the time is spent between 0 and 4000 ft. |
| As applicable (except II-e, 3, 4, 9, 10 and II 1, 3, 4, 7, 8) | Gross weight | 2400 lb ~ 100% | Reference 1, Figure 5 shows over two-thirds of time spent below 2400 lb. Not practical to expand further and add additional points to spectrum. |
| I-a | rpm | Maximum ~ 100% | Recommended flight manual procedure for takeoff. |
| II-a, II-f, II-k | Control motion | ±25% ~ 100% | Based on design data. The Reference 1 data which shows cyclic peaks of 40% control motion is inconclusive in that the maximum is not given. |
| II-a, II-b, II-j, II-k | Reversal and hover turn rate | Rapid ~ 70% Slow ~ 30% | Based on design data. |
| *II-c, II-l | Airspeed | Per Figure 2 | Airspeed data from Reference 1, Figure 11.b. |
| *II-c, II-e, II-j, II-l | Load factor | Per Table V | Based on load factor exceedance data of Reference 1, Figure 12.c. |
| *II-e 3, 4, 9, 10 II-l 3, 4, 7, 8 | Gross weight | 2200 lb ~ 100% | Reference 1 shows maximum g maneuvers at reduced gross weight. |
| II-g, IV-c | Density altitude | 3000 ft $H_d$ ~ 80% 7000 ft $H_d$ ~ 15% 11000 ft $H_d$ ~ 5% | The design spectrum breakdown shown at left yields approximately 9% of occurrence at > 4000 ft which is conservative compared to the altitude data of Reference 1, Figure 7. |
| *II-h | Load factor | Minimum g (0. 2) ~ 100% | To cover load factor exceedance data of Reference 1, Figure 12.c. |
| *II-j, II-k, II-l, Iv-e | Autorotation rpm | Maximum ~ 90% Minimum ~ 10% | Bulk of time (84%) spent at high gross weight (> 2200 lb) which requires high autorotation rpm. Also, Reference 1, Figure 6 shows 3% of time at rpm > 490, only 0. 1% of time at rpm < 460. |
| *III-a | Airspeed | 30 knots ~ 90% 0 knots ~ 10% | Determined to be more realistic than design breakdown by pilots during Hughes developmental tests. |
| III-b | rpm | Maximum ~ 66-2/3% Minimum ~ 33-1/3% | Higher rpm recommended for this type of operation due to added safety margin is weighted more heavily. |
| *IV-d | Airspeed | 5 mph ~ 50% 10 mph ~ 24% 15 mph ~ 14% 20 mph ~ 8% 25 mph ~ 4% | Determined to be more realistic than design condition for simulation of hovering in sidewind conditions. |
| IV-e | Percent Occurrence | Zero | Conditions so noted are demonstration points required by the FAA. |

*Indicates detailed condition distribution different than corresponding design spectrum detailed condition distribution.

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TABLE IV. AR-56 SPECTRUM — CONVERSION TO BASIC CONDITIONS PERCENTAGE
OF OCCURRENCE (Data From Reference 6)

### A. MANEUVER PER 100 FLIGHT HOURS CONDITIONS

| Condition | Maneuvers/ 100 Hours | Maneuver Duration-Seconds* | Time-Seconds | Percentage of ** Occurrence |
|---|---|---|---|---|
| Takeoff | 400 | 10 | 4,000 | 0.98 |
| Turns Hovering | 1,000 | 6 | 6,000 | 1.47 |
| Control Reversals Hovering | 1,000 | 3 | 3,000 | 0.73 |
| Landing Approach | 500 | 25 | 12,500 | 3.06 |
| Partial Power Descent | 500 | 40 | 20,000 | 4.89 |
| Control Reversals | 800 | 3 | 2,400 | 0.59 |
| Pullups | 250 | 3 | 750 | 0.18 |
| Power to Autorotation | 40 | 3 | 120 | 0.03 |
| Autorotation to Power | 40 | 3 | 120 | 0.03 |
| Auto. - Pullups | 40 | 3 | 120 | 0.03 |
| | | | 49,010 | 11.99 Total A |

49,010 seconds = 13.6 hours (from A)
Add 100 hours (from B) = 113.6 hours, combined time percentage of occurrence basis

### B. PERCENTAGE OF SERVICE LIFE CONDITIONS

| Condition | Percent | Percentage of *** Occurrence |
|---|---|---|
| Ground | 1.0 | 0.88 |
| Hovering | 10.0 | 8.80 |
| Sideward | 1.0 | 0.88 |
| Rearward | 0.5 | 0.44 |
| 20% $V_H$ | 5.0 | 4.40 |
| 40% $V_H$ | 5.0 | 4.40 |
| 50% $V_H$ | 2.0 | 1.76 |
| 60% $V_H$ | 8.0 | 7.04 |
| 70% $V_H$ | 10.0 | 8.80 |
| 80% $V_H$ | 15.0 | 13.20 |
| 90% $V_H$ | 18.0 | 15.85 |
| 100% $V_H$ | 10.0 | 8.80 |
| 115% $V_H$ | 1.0 | 0.88 |
| Takeoff Power Climb | 1.0 | 0.88 |
| Full Power Climb | 3.0 | 2.64 |
| Dives | 2.5 | 2.20 |
| Right Turns | 2.5 | 2.20 |
| Left Turns | 2.5 | 2.20 |
| Autorotation | 1.0 | 0.88 |
| Auto. - Left Turn | 0.2 | 0.18 |
| Auto. - Right Turn | 0.2 | 0.18 |
| Auto. - Control Reversals | 0.3 | 0.26 |
| Auto. Landing | 0.3 | 0.26 |
| | 100.0 | 88.01 Total B |

*Same as Hughes design data
**Based on Time/Combined Time
***Ratio 100/113.6

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whereas the AMCP 706-203 spectrum shows the poorest comparison. The time spent in the maneuver mission segment during the latter is considerably lower than for the other spectra.

The most notable difference between the operational spectrum and the others, from a mission segment standpoint, is the greater percentage of time spent in maneuvers (and less in steady state) operationally than predicted by the three analytical profiles.

## Basic Conditions

The operational spectrum contains the most complete listing of basic conditions as reflected by current Hughes mission profile knowledge. Several basic conditions have been added to the operational spectrum which do not appear in the original design spectrum presented. The basic conditions are hovering turns, pushovers, approach to hover and hover OGE.

The basic condition definitions from the AR-56 (Reference 6) spectrum match up well against the common definitions for the operational and design spectrums. Basic conditions not accounted for in the AR-56 spectrum are pushovers and simulated power failures.

The spectrum based on AMCP 706-203 (Reference 5) excludes several basic conditions that should be part of a complete mission profile. The conditions are simulated power failure, and turns, pullups and control reversals in autorotation. The terminology used for basic condition identification would seem to be more indicative of a combat vehicle, but such is not the case as was shown previously by the comparatively low amount of time spent in the maneuver mission segment.

Other than the basic condition deletions already mentioned, the most significant differences between the mission profiles from a basic condition percentage of occurrence standpoint are as follows:

1.  The AR-56 and AMCP 706-203 spectra show an extremely low amount of time spent in autoratation.

2.  The hover condition in the AR-56 spectrum appears to have an excessively large percentage of occurrence.

3.  Too much time is given to the ground conditions (that is, flat pitch, start and shutdown) in the AMCP 706-203 spectrum.

As discussed during the derivation of the operational spectrum, the air-speed breakdown used for the operational steady-state mission segment basic conditions is shown in Figure 1. The design spectrum airspeeds employed in the steady-state segment were converted to the same format and added to the graph. A comparison of the percentage of time in each airspeed range is available from the figure. The data show that more time is spent in the operational spectrum in the $V_H$ and 80 percent $V_H$ ranges than for the design spectrum. However, the design spectrum actually matches up quite well against the data taken directly from Reference 1 before specific airspeed ranges were identified for operational spectrum definition.

### Detailed Conditions

Differences in the detailed conditions and percentage of occurrences between the design and operational spectra shown in Table II were brought about for two reasons. Either Hughes gained new knowledge during OH-6A developmental testing that justified revising the original OH-6A design spectrum as presented herein for future programs, or operational data included in Reference 1 superceded the design data. The operational detailed condition percentage of occurrence distributions, which differ from the corresponding design detailed condition distributions, are noted by an asterisk in Table III. The two major differences are discussed below.

A comparison of maneuver mission segment airspeeds for the operational and design spectrums is presented in Figure 2. The design airspeed figures are quite conservative since the bulk of operational spectrum maneuver segment time is conducted at considerably lower airspeeds than required by the design mission profile program. The conservatism of the design spectrum is even more pronounced compared to the cumulative airspeed data obtained directly from Reference 1.

Table V presents the operational spectrum load factor occurrence based on the operational maneuver mission segment data. Also shown is the design spectrum load factor occurrence for the same mission segment. The design data are based on the percentage of occurrences for turns and pullups, and the buildup and maximum g load factors that were assigned during the OH-6A flight testing of the detailed conditions spectrum. Both the number of occurrences and the load factor level of the design spectrum are conservative when compared to the operational data of Reference 1. The number of occurrences for the operational spectrum exceeds the design values due to the basic condition percentage of occurrence calculations which were discussed earlier in this task.
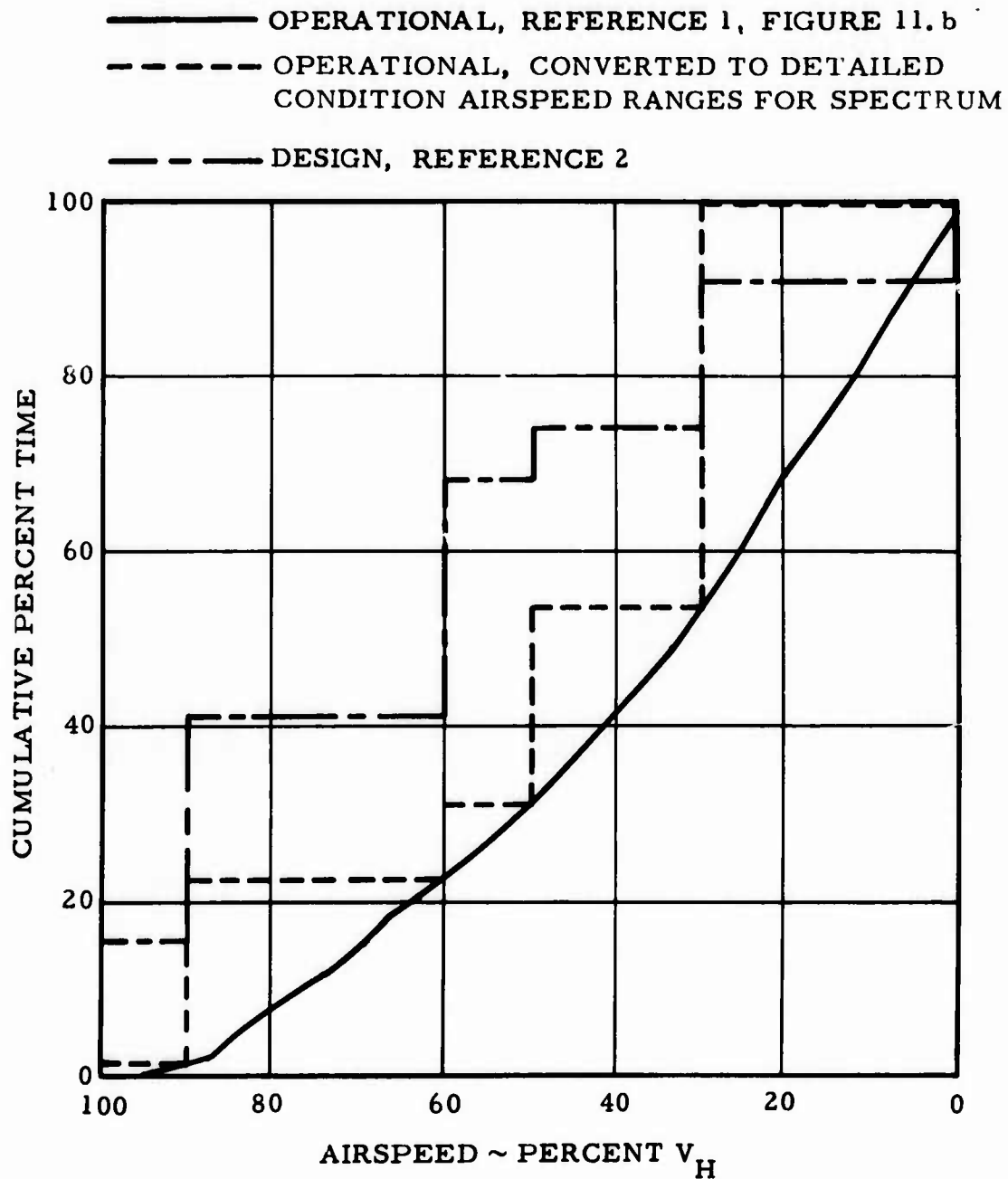
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Figure 2. Cumulative Airspeed Data - Maneuver Mission Segment.

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# TABLE V. LOAD FACTOR OCCURRENCE – MANEUVER MISSION SEGMENT

**Operational Data (Reference 1, Figure 12.c) (Corrected to Composite)**

| Load Factor ~g | Occurrences in 1,000 Hours |
|---|---|
| 1.3 to 1.39 | 8,500 |
| 1.4 to 1.49 | 6,800 |
| 1.5 to 1.59 | 5,100 |
| 1.6 to 1.79 | 4,371 |
| 1.8 to 1.99 | 641 |
| 2.0 to 2.19 | 78 |
| > 2.19 | 10 |
| Total | 25,500 |

**Operational Spectrum [From Table II] (Turns and Pullups Only)**

| | Load Factor | Occurrences in 1,000 Hours |
|---|---|---|
| Autorotation | Buildup g (1.5) | 25,544 |
| Power-on** | Buildup g (1.8) | 56,744 |
| Turns | Maximum g (2.06) | 1,026 |
| Pullups | Maximum g (2.20) | 413 |
| | | 83,727 |

NOTE: Load factor occurrences less than 1.3 g are included in the steady-state mission segment (conditions IV-c 19 and 20) which also accounts for the gust load factor occurrences between 1.2 and 1.3 g shown in Reference 1, Figure 15.

**Design Spectrum (Reference 1) (Turns and Pullups Only)**

| | Load Factor ~g | Occurrences in 1,000 Hours |
|---|---|---|
| Turns | 2.00 | 35,380 |
| Pullups | 2.00 | 22,380 |
| Turns | 2.06* | 12,420 |
| Pullups | 2.55* | 1,620 |
| | | 72,000 |

NOTE: Refer to Task IV for a discussion relating the load factors shown above to those actually achieved during OH-6A developmental programs.

*Values shown are for 2400 lb gross weight. Similar data for 2100 lb can be obtained by multiplying load factors shown by 2400/2100.

**Except for Conditions II-c 9, 10, 23, 24 and II-e 5, 6 which are 1.5 g.

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# TASK II - MAIN AND TAIL ROTOR FATIGUE DATA COMPARISON

## INTRODUCTION

Flapwise bending of the m̲in rotor (M/R) blade at the 15 percent radius is the structural load measurement that determines the fatigue life of the main rotor blade on the OH-6A Helicopter. Likewise, the resultant of flapwise and chordwise bending moments measured at the 7-inch radius on the tail rotor (T/R) determines the fatigue life of the tail rotor blade. These two fatigue load measurements were selected for study in this task. The trends evident in a study of these load parameters are indicative of trends for other loads in the main and tail rotor systems, and hence, reflect the effect of mission profiles on rotor system component fatigue data. Fatigue load spectra, damage rates and service lives are presented as part of the analysis.

## FATIGUE LOAD SPECTRA

### Design Mission Profile Fatigue Load Spectra

Fatigue load spectra for the main and tail rotor blades of the OH-6A helicopter used during engineering development testing are available in OH-6A flight strain surveys (References 7 and 8). The data are presented for each design spectrum detailed condition in Table VI. The design spectrum detailed conditions were identified from Task I (Table II) as those showing a value under the design percentage of occurrence column. The detailed condition number shown in the first column of Table VI corresponds to the numbering system used in the detailed condition listing (Table II). Histograms of load versus frequency of occurrence for the design spectrum using the data from the aforementioned tables are shown in Figures 3 and 4 for the main and tail rotor, respectively.

### Operational Mission Profile Fatigue Load Spectra

In order to define the operational mission profile fatigue load spectra, main and tail rotor loads had to be determined for each detailed test condition showing an operational percentage of occurrence in Table II. Inasmuch as many of the operational mission profile detailed conditions match the design detailed conditions, the loads shown in Table VI for those conditions are directly applicable to both mission profiles. However, there

# TABLE VI.  FATIGUE LOAD SPECTRA
(Refer to Table II for Percentage of Occurrence)

| Detailed Condition Number | (1) ± inch-pounds | (2) ± inch-pounds | Detailed Condition Number | (1) ± inch-pounds | (2) ± inch-pounds | Detailed Condition Number | (1) ± inch-pounds | (2) ± inch-pounds |
|---|---|---|---|---|---|---|---|---|
| I-a 1 | 258 | 286 | II-c 27 | 1665, 1910 | 565, N/A | II-j 7 | 498, 290 | 264, N/A |
| I-b 1 | 283 | 225 | II-c 28 | 1482, 2310 | 441, N/A | II-j 8 | 368, 1280 | 224, N/A |
| I-b 2 | 385 | 322 | II-d 1 | 222 | 216 | II-j 9 | 410 | N/A |
| I-b 3 | 254 | 176 | II-d 2 | 178 | 236 | II-j 10 | 1200 | N/A |
| I-b 4 | 275 | 284 | II-d 3 | 445 | 233 | II-j 11 | 444, 1160 | 505, N/A |
| II-a 1 | 215 | II-a 4 | II-d 4 | 222 | 327 | II-j 12 | 1135, 1970 | 280, N/A |
| II-a 2 | 196 | II-a 4 | II-d 5 | 529 | 562 | II-j 13 | 250 | N/A |
| II-a 3 | 182 | 218 | II-d 6 | 600 | 397 | II-j 14 | 250 | N/A |
| II-a 4 | 223 | 286 | II-e 1 | 871, 280 | 260, N/A | II-j 15 | 283, 250 | 199, N/A |
| II-a 5 | 196 | II-a 8 | II-e 2 | 1243, 280 | 649, N/A | II-j 16 | 366, 250 | 274, N/A |
| II-a 6 | 313 | II-a 8 | II-e 3 | 280 | N/A | II-j 17 | 290 | N/A |
| II-a 7 | 263 | 224 | II-e 4 | 440 | N/A | II-j 18 | 310 | N/A |
| II-a 8 | 284 | 257 | II-e 5 | 610 | N/A | II-j 19 | 476, 290 | 194, N/A |
| II-a 9 | 179 | 724 | II-e 6 | 940 | N/A | II-j 20 | 390, 1280 | 309, N/A |
| II-a 10 | 197 | 702 | II-e 7 | 503, 1320 | 482, N/A | II-j 21 | 410 | N/A |
| II-a 11 | 223 | 504 | II-e 8 | 1143, 1680 | 371, N/A | II-j 22 | 1200 | N/A |
| II-a 12 | 101 | 567 | II-e 9 | 2420, 1820 | 781, N/A | II-j 23 | 550, 1160 | 474, N/A |
| II-b 1 | 285 | 863 | II-e 10 | 1063, 2210 | 339, N/A | II-j 24 | 862, 1970 | 427, N/A |
| II-b 2 | II-b 1 | II-b 1 | II-f 1 | 1463 | 558 | II-k 1 | 469, 319 | 297, 163 |
| II-b 3 | 271 | 658 | II-f 2 | 762 | 611 | II-k 2 | 1359, 924 | 647, 356 |
| II-b 4 | II-b 3 | II-b 3 | II-f 3 | 800 | II-f 4 | II-k 3 | 1074, 730 | 246, 135 |
| II-c 1 | 589, 250 | 232, N/A | II-f 4 | 1580 | 353 | II-k 4 | 1438, 978 | II-k 3 |
| II-c 2 | 387, 250 | 173, N/A | II-f 5 | 704 | 606 | II-k 5 | 469, 319 | 350, 193 |
| II-c 3 | 250 | N/A | II-f 6 | 998 | 667 | II-k 6 | 1152, 783 | 571, 314 |
| II-c 4 | 250 | N/A | II-f 7 | 832 | II-f 8 | II-k 7 | 725, 493 | 153, 84 |
| II-c 5 | 677, 300 | 329, N/A | II-f 8 | 1503 | 300 | II-k 8 | 1042, 709 | II-k 7 |
| II-c 6 | 487, 360 | 264, N/A | II-f 9 | 862 | 879 | II-k 9 | 346 | 443 |
| II-c 7 | 620 | N/A | II-f 10 | 674 | 788 | II-k 10 | 486 | 662 |
| II-c 8 | 990 | N/A | II-f 11 | 871 | 532 | II-k 11 | 285 | 460 |
| II-c 9 | 610 | N/A | II-f 12 | 1083 | 838 | II-k 12 | 549 | 1037 |
| II-c 10 | 940 | N/A | II-g 1 | 913 | 520 | II-k 13 | 407 | 132 |
| II-c 11 | 801, 1320 | 307, N/A | II-g 2 | 508 | 549 | II-k 14 | 506 | 466 |
| II-c 12 | 433, 1680 | 384, N/A | II-g 3 | 377 | 478 | II-k 15 | 305 | 306 |
| II-c 13 | 1671, 1910 | 478, N/A | II-h 1 | 236 | 338 | II-k 16 | 1078 | 576 |
| II-c 14 | 1585, 2310 | 557, N/A | II-h 2 | II-h 1 | II-h 1 | II-l 1 | 270 | N/A |
| II-c 15 | 492, 250 | 187, N/A | II-h 3 | 405 | 377 | II-l 2 | 280 | N/A |
| II-c 16 | 408, 250 | 284, N/A | II-h 4 | II-h 3 | II-h 3 | II-l 3 | 270 | N/A |
| II-c 17 | 250 | N/A | II-h 5 | II-h 3 | II-h 3 | II-l 4 | 710 | N/A |
| II-c 18 | 250 | N/A | II-h 6 | II-h 3 | II-h 3 | II-l 5 | 553, 410 | 242, N/A |
| II-c 19 | 679, 300 | 318, N/A | II-i 1 | 285 | 250 | II-l 6 | 298, 1200 | 226, N/A |
| II-c 20 | 509, 360 | 269, N/A | II-i 2 | 366 | 278 | II-l 7 | 553, 1070 | 242, N/A |
| II-c 21 | 620 | N/A | II-j 1 | 250 | N/A | II-l 8 | 298, 1570 | 226, N/A |
| II-c 22 | 990 | N/A | II-j 2 | 250 | N/A | III-a 1 | 1266 | 282 |
| II-c 23 | 610 | N/A | II-j 3 | 381, 250 | 336, N/A | III-a 2 | 1268 | 270 |
| II-c 24 | 940 | N/A | II-j 4 | 305, 250 | 215, N/A | III-a 3 | 469 | 580 |
| II-c 25 | 390, 1320 | 393, N/A | II-j 5 | 290 | N/A | III-a 4 | II-a 3 | III-a 3 |
| II-c 26 | 606, 1680 | 497, N/A | II-j 6 | 310 | N/A | III-a 5 | 1689 | III-a 3 |

| TABLE VI — Continued | | | | | | | | |
|---|---|---|---|---|---|---|---|---|
| Detailed Condition Numbers | (1) ± inch-pounds | (2) ± inch-pounds | Condition Number | (1) ± inch-pounds | (2) ± inch-pounds | Detailed Condition Number | ± inch-pounds | ± inch-pounds |
| III-a 6 | III-a 5 | III-a 3 | IV-c 21 | 368 | 419 | IV-d 13 | N/A* | N/A |
| III-b 1 | 952 | 244 | IV-c 22 | 247 | 341 | IV-d 14 | N/A* | N/A |
| III-b 2 | 790 | 330 | IV-c 23 | 305 | 310 | IV-d 15 | N/A** | N/A |
| III-b 3 | 1179 | 654 | IV-c 24 | 308 | 200 | IV-d 16 | N/A** | N/A |
| III-b 4 | III-b 3 | III-b 3 | IV-c 25 | IV-c 31 | IV-c 31 | IV-d 17 | N/A** | N/A |
| III-b 5 | 786 | 775 | IV-c 26 | IV-c 32 | IV-c 32 | IV-d 18 | N/A** | N/A |
| III-b 6 | III-b 5 | III-b 5 | IV-c 27 | IV-c 33 | IV-c 33 | IV-d 19 | N/A** | N/A |
| III-c 1 | 860 | 492 | IV-c 28 | IV-c 34 | IV-c 34 | IV-d 20 | N/A** | N/A |
| IV-a 1 | 386 | 398 | IV-c 29 | IV-c 35 | IV-c 35 | IV-d 21 | N/A** | N/A |
| IV-a 2 | 390 | 238 | IV-c 30 | IV-c 36 | IV-c 36 | IV-d 22 | N/A** | N/A |
| IV-b 1 | 193 | 209 | IV-c 31 | 609 | 452 | IV-d 23 | N/A** | N/A |
| IV-b 2 | 218 | 248 | IV-c 32 | 544 | 426 | IV-d 24 | N/A** | N/A |
| IV-b 3 | N/A | N/A | IV-c 33 | 599 | 372 | IV-d 25 | 309 | 366 |
| IV-b 4 | N/A | N/A | IV-c 34 | 637 | 423 | IV-d 26 | 243 | 224 |
| IV-c 1 | 516 | 186 | IV-c 35 | 397 | 312 | IV-d 27 | N/A*** | N/A |
| IV-c 2 | 450 | 332 | IV-c 36 | 835 | 325 | IV-d 28 | N/A*** | N/A |
| IV-c 3 | 327 | 200 | IV-c 37 | 810 | 498 | IV-d 29 | N/A*** | N/A |
| IV-c 4 | 577 | 250 | IV-c 38 | 1009 | 519 | IV-d 30 | N/A*** | N/A |
| IV-c 5 | 305 | 204 | IV-c 39 | 799 | 490 | IV-d 31 | N/A*** | N/A |
| IV-c 6 | 637 | 233 | IV-c 40 | 1280 | 488 | IV-d 32 | N/A*** | N/A |
| IV-c 7 | 275 | 249 | IV-c 41 | 956 | 375 | IV-d 33 | N/A*** | N/A |
| IV-c 8 | 222 | 298 | IV-c 42 | 1321 | 413 | IV-d 34 | N/A*** | N/A |
| IV-c 9 | 338 | 233 | IV-d 1 | 175 | 334 | IV-d 35 | N/A*** | N/A |
| IV-c 10 | 348 | 260 | IV-d 2 | 142 | 273 | IV-d 36 | N/A*** | N/A |
| IV-c 11 | 244 | 347 | IV-d 3 | 143 | 381 | IV-e 1 | 468 | 185 |
| IV-c 12 | 391 | 183 | IV-d 4 | 182 | 401 | IV-e 2 | 402 | 396 |
| IV-c 13 | 319 | 239 | IV-d 5 | N/A* | N/A | IV-e 6 | 563 | 479 |
| IV-c 14 | 378 | 378 | IV-d 6 | N/A* | N/A | IV-e 8 | 478 | 304 |
| IV-c 15 | 284 | 378 | IV-d 7 | N/A* | N/A | IV-e 13 | 423 | 271 |
| IV-c 16 | 307 | 287 | IV-d 8 | N/A* | N/A | IV-e 18 | 233 | 169 |
| IV-c 17 | 288 | 125 | IV-d 9 | N/A* | N/A | IV-e 19 | 324 | 191 |
| IV-c 18 | 391 | 208 | IV-d 10 | N/A* | N/A | IV-e 20 | 721 | 341 |
| IV-c 19 | 339, 380 | 286, N/A | IV-d 11 | N/A* | N/A | | | |
| IV-c 20 | 523, 400 | 486, N/A | IV-d 12 | N/A* | N/A | | | |

NOTE: (1) Cyclic main rotor blade 15% radius flapwise bending moment.

(2) Cyclic tail rotor 7-inch radius flapwise and chordwise bending moment (resultant).

(3) For case of two values shown, first applies to design spectrum, second to operational spectrum.

(4) Where a detailed condition number appears in a load space, use the load corresponding to that detailed number.

(5) N/A = not available. Refer to Table VII for explanation.

*Use maximum of conditions IV-d 1, 2
**Use maximum of conditions IV-d 3, 4
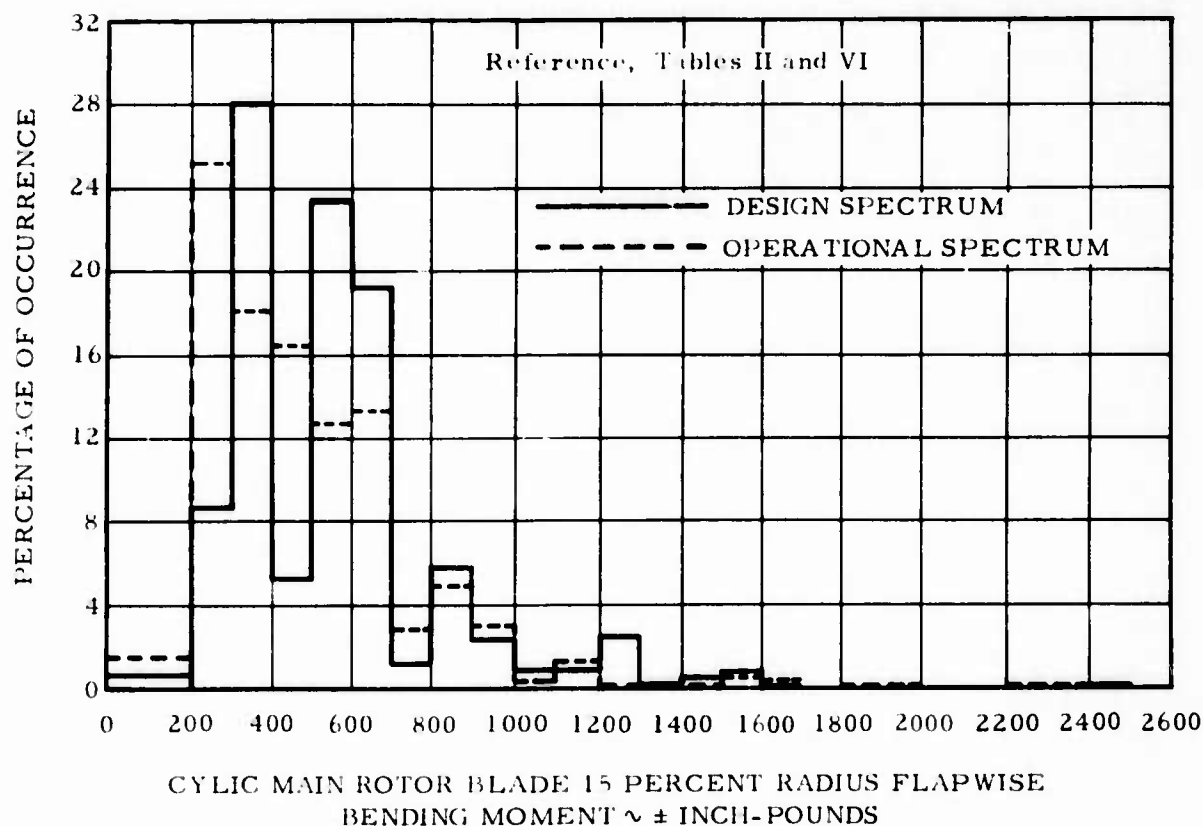***Use maximum of conditions IV-d 25, 26

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Figure 3. Fatigue Load Versus Percentage of Occurrence - Main Rotor.

are several operational detailed conditions that differ in airspeed and/or load factor requirements from the corresponding design condition. Additionally, some operational conditions do not appear in the design mission profile. Table VII lists the source used to obtain the fatigue load for all operational conditions that do not match a design condition. The loads data were obtained by three methods:

1. From Hughes flight test programs other than the OH-6A design mission profile testing.

2. By application of a factor based on design spectrum data (Table VII, Item 6).

3. From the analysis described in the following discussion. (This analysis is applicable to the operational conditions II. c, II. e, II. j, II. l, and part of IV. c, which differ from design conditions in airspeed or load factor requirements.)

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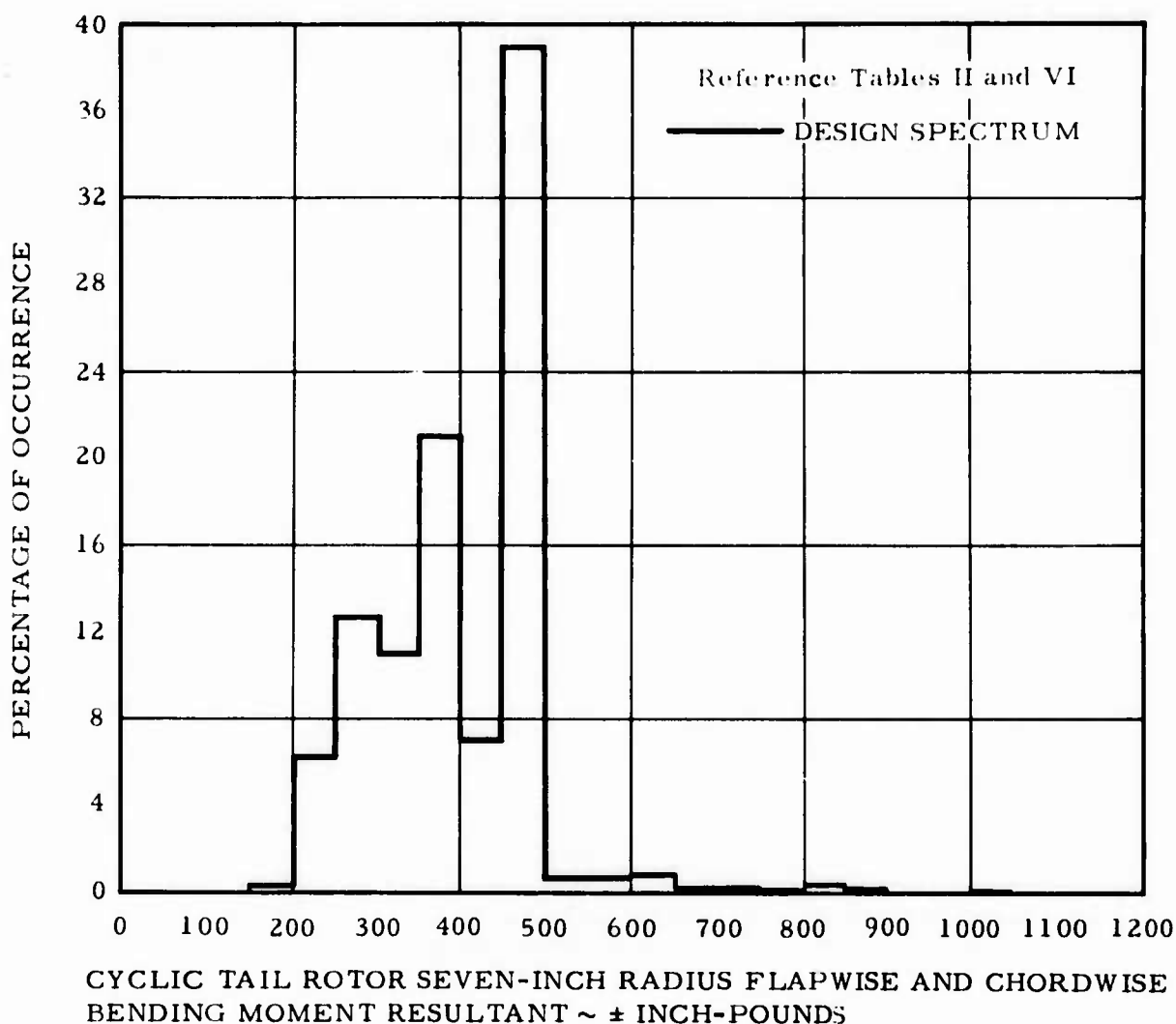
Figure 4. Fatigue Load Versus Percentage of Occurrence - Tail Rotor.

In order to develop the operational main rotor fatigue load spectra for these conditions, it was necessary to establish a relationship between the magnitude of the design mission profile fatigue load measurements and some generalized parameter that could be defined by a combination of the elements of either mission profile. Table VIII presents main rotor fatigue load data selected from OH-6A flight strain surveys along with $\mu$ (forward speed divided by rotor tip speed) and $C_T/\sigma$ (thrust/$A_b \rho V_T^2$ where thrust is gross weight times load factor). These design data points were converted to graphical form (Figure 5) where 15 percent flapwise bending moment of the main rotor blade is plotted against $\mu$ with trend lines for $C_T/\sigma$.

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| | | TABLE VII. OPERATIONAL SPECTRUM — FATIGUE LOAD DETERMINATION | |
|---|---|---|---|
| Item | Detailed Condition Number | Main Rotor* | Tail Rotor* |
| 1 | II-b 1 thru II-b 4 | Test program reported in Reference 9 | Test program reported in Reference 9 |
| 2 | II-c 1 thru II-c 28 | Figure 5 | N/A |
| 3 | II-e 1 thru II-e 10 | Figure 5 | N/A |
| 4 | II-h 1 thru II-h 6 | Reference 10 | Reference 10 |
| 5 | II-j 1 thru II-j 24 | Figure 5 | N/A |
| 6 | II-k 1 thru II-k 8 | An average ratio was obtained from conditions II-k 9, II-k 16 which show the effect on autorotation control reversal loads for a reduction in airspeed from $V_{NE}$ to $0.5\ V_{NE}$. The ratio was applied to the design spectrum loads. | |
| 7 | II-l 1 thru II-l 8 | Figure 5 | N/A |
| 8 | III-a 3 thru III-a 6 | Reference 11 | Reference 9 |
| 9 | III-b 3 thru III-b 6 | Reference 11 | Reference 9 |
| 10 | IV-b 3 thru IV-b 4 | N/A | N/A |
| 11 | IV-c 19 and IV-c 20 | Figure 5 | N/A |
| 12 | IV-d 5 thru IV-d 24, IV-d 27 thru IV-d 36 | In order to complete Figure 3 (Fatigue Load Spectra Histogram), design spectrum data, which is conservative, will be used. See Table VI. | N/A |

NOTE: Conditions marked as N/A (not available) are also shown that way in Table VI. Neither flight test data nor an analytical means is available to determine the loads. However, for items 2, 3, 5, 7 and 11 the operational detailed conditions are the same as the design conditions except for airspeed and/or load factor. Inasmuch as the design conditions do not cause fatigue damage to the tail rotor, the same assumption is made for the operational conditions. Also, Hughes experience has shown that flight test conditions similar to the operational conditions of items 10 and 12 do not affect the service lives of either the main or tail rotor.

*Fatigue load shown in Table VI obtained from the source noted.

# TABLE VIII. FATIGUE LOAD VERSUS μ AND $C_T/\sigma$
## (References 7 and 8 Level Flight, Pullups and Turns)

| Detailed Condition Number | (1) ± inch-pounds | μ ~ $V_f/V_T$ | $C_T/\sigma$ ~ $\dfrac{\text{Thrust}}{A_b \rho V_T^2}$ | Detailed Condition Number | (1) ± inch-pounds | μ ~ $V_f/V_T$ | $C_T/\sigma$ ~ $\dfrac{\text{Thrust}}{A_b \rho V_T^2}$ |
|---|---|---|---|---|---|---|---|
| IV-c 13 | 319 | 0.197 | 0.089 | II-e 10 | 1063 | 0.321 | 0.137 |
| IV-c 14 | 378 | 0.244 | 0.127 | IV-c 13 | 163 | 0.209 | 0.081 |
| IV-c 19 | 339 | 0.270 | 0.092 | IV-c 14 | 218 | 0.218 | 0.087 |
| IV-c 20 | 523 | 0.320 | 0.122 | IV-c 19 | 272 | 0.261 | 0.080 |
| IV-c 31 | 609 | 0.341 | 0.088 | IV-c 20 | 294 | 0.269 | 0.083 |
| IV-c 32 | 544 | 0.358 | 0.088 | IV-c 31 | 479 | 0.316 | 0.080 |
| II-j 3 | 381 | 0.104 | 0.078 | IV-c 31 | 441 | 0.318 | 0.084 |
| II-j 15 | 283 | 0.110 | 0.070 | IV-c 31 | 482 | 0.321 | 0.087 |
| II-j 4 | 305 | 0.134 | 0.103 | IV-c 31 | 493 | 0.316 | 0.079 |
| II-j 16 | 366 | 0.135 | 0.107 | IV-c 31 | 559 | 0.335 | 0.069 |
| II-j 7 | 498 | 0.184 | 0.067 | IV-c 32 | 359 | 0.296 | 0.083 |
| II-j 19 | 476 | 0.187 | 0.090 | IV-c 32 | 431 | 0.319 | 0.093 |
| II-j 8 | 368 | 0.251 | 0.103 | IV-c 32 | 242 | 0.304 | 0.098 |
| II-j 20 | 390 | 0.249 | 0.118 | IV-c 32 | 360 | 0.305 | 0.088 |
| II-j 11 | 444 | 0.294 | 0.103 | IV-c 32 | 437 | 0.317 | 0.072 |
| II-j 23 | 550 | 0.284 | 0.058 | II-j 23 | 365 | 0.242 | 0.069 |
| II-j 12 | 1135 | 0.354 | 0.160 | IV-c 15 | 185 | 0.153 | 0.088 |
| II-j 24 | 862 | 0.366 | 0.195 | IV-c 15 | 284 | 0.167 | 0.094 |
| IV-c 15 | 267 | 0.118 | 0.094 | IV-c 16 | 240 | 0.158 | 0.096 |
| IV-c 16 | 307 | 0.155 | 0.107 | IV-c 21 | 185 | 0.208 | 0.088 |
| IV-c 21 | 206 | 0.220 | 0.101 | IV-c 21 | 368 | 0.222 | 0.092 |
| IV-c 22 | 247 | 0.224 | 0.111 | IV-c 22 | 207 | 0.215 | 0.095 |
| IV-c 33 | 447 | 0.304 | 0.096 | IV-c 33 | 272 | 0.294 | 0.090 |
| IV-c 34 | 469 | 0.323 | 0.098 | IV-c 33 | 478 | 0.290 | 0.109 |
| IV-c 17 | 288 | 0.144 | 0.108 | IV-c 33 | 599 | 0.278 | 0.091 |
| IV-c 18 | 391 | 0.145 | 0.118 | IV-c 33 | 514 | 0.294 | 0.086 |
| IV-c 23 | 305 | 0.159 | 0.132 | IV-c 34 | 283 | 0.275 | 0.097 |
| IV-c 24 | 308 | 0.176 | 0.121 | IV-c 34 | 468 | 0.289 | 0.103 |
| IV-c 35 | 397 | 0.262 | 0.094 | IV-c 34 | 468 | 0.273 | 0.111 |
| II-c 1 | 589 | 0.124 | 0.123 | IV-c 34 | 473 | 0.275 | 0.105 |
| II-c 2 | 387 | 0.160 | 0.158 | IV-c 34 | 637 | 0.274 | 0.098 |
| II-c 15 | 492 | 0.143 | 0.141 | IV-c 35 | 344 | 0.241 | 0.106 |
| II-c 16 | 408 | 0.150 | 0.159 | IV-c 36 | 312 | 0.245 | 0.114 |
| II-c 5 | 677 | 0.191 | 0.165 | IV-c 5 | 494 | 0.191 | 0.101 |
| II-c 6 | 487 | 0.210 | 0.120 | IV-c 17 | 515 | 0.187 | 0.112 |
| II-c 19 | 879 | 0.194 | 0.153 | II-c 13 | 1084 | 0.284 | 0.148 |
| II-c 20 | 509 | 0.211 | 0.123 | II-c 13 | 1671 | 0.316 | 0.145 |
| II-c 11 | 801 | 0.340 | 0.119 | II-c 13 | 1527 | 0.310 | 0.140 |
| II-c 12 | 433 | 0.346 | 0.110 | II-c 13 | 1127 | 0.293 | 0.137 |
| II-c 25 | 390 | 0.352 | 0.106 | II-c 14 | 977 | 0.266 | 0.143 |
| II-c 26 | 606 | 0.348 | 0.118 | II-c 14 | 1114 | 0.296 | 0.144 |
| II-e 1 | 871 | 0.256 | 0.146 | II-c 14 | 853 | 0.287 | 0.138 |
| II-e 2 | 1243 | 0.286 | 0.152 | II-c 14 | 1585 | 0.317 | 0.137 |
| II-e 7 | 503 | 0.307 | 0.109 | II-c 27 | 1014 | 0.288 | 0.164 |
| II-e 8 | 1143 | 0.305 | 0.120 | II-c 27 | 914 | 0.318 | 0.126 |
| II-l 5 | 747 | 0.288 | 0.154 | II-c 27 | 1665 | 0.316 | 0.123 |
| II-l 6 | 1342 | 0.347 | 0.167 | II-c 27 | 1006 | 0.294 | 0.117 |
| II-c 13 | 1671 | 0.301 | 0.154 | II-c 28 | 1345 | 0.290 | 0.182 |
| II-c 14 | 978 | 0.299 | 0.138 | II-c 28 | 1482 | 0.296 | 0.166 |
| II-c 27 | 1614 | 0.307 | 0.141 | II-c 28 | 691 | 0.281 | 0.143 |
| II-c 28 | 1211 | 0.322 | 0.141 | II-c 28 | 965 | 0.317 | 0.134 |
| II-e 9 | 2420 | 0.316 | 0.168 | II-e 9 | 1775 | 0.323 | 0.144 |

NOTES:  (1) Cyclic main rotor blade 15% radius flapwise bending moment.
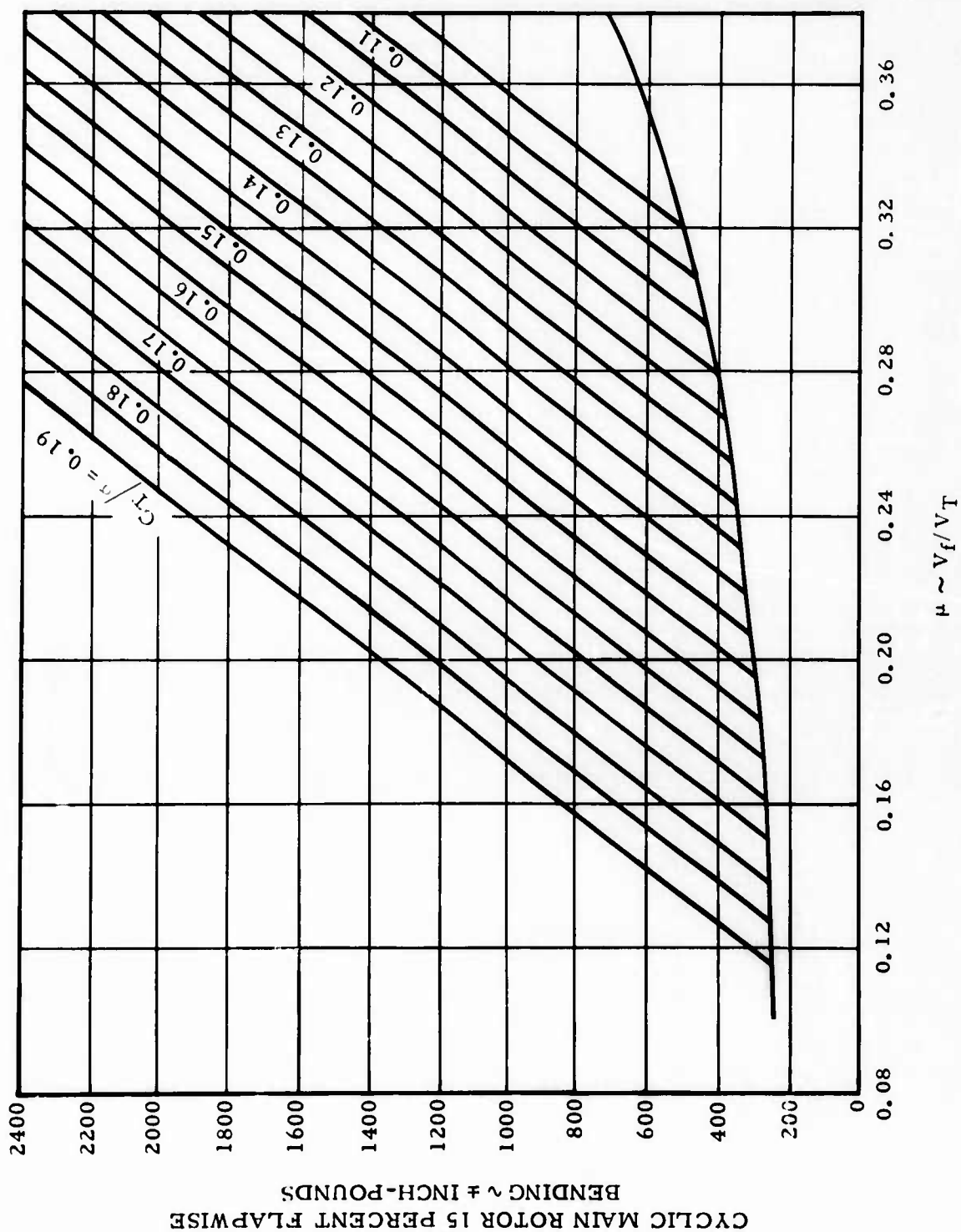(2) $A_b$ = 29.625 ft$^2$ = main rotor blade area.

Figure 5. Fatigue Load Versus μ - Main Rotor.

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Table IX presents a list of the operational spectrum detailed conditions for which the main rotor fatigue load had to be determined. The calculated $C_T/\sigma$ and $\mu$ values based on the detailed condition definitions are also shown. Knowing these values, Figure 5 was entered at the appropriate $\mu$ and $C_T/\sigma$ point, and interpolating as necessary, the main rotor fatigue load was read from the ordinate. The data are shown for each detailed condition in Table IX.

With the addition of the above data to Table VI, the main and tail rotor fatigue load spectra were completed for the operational mission profile as well as the design mission profile. Using the operational spectrum percentage of occurrences, the operational main rotor data were added to the histogram (Figure 3) for comparison with the design mission profile.

Because of the number of conditions and percentage of occurrence for which tail rotor fatigue loads were not actually determined (see note Table VII), and since a complete histogram comparison would not be available, the operational data are not included in Figure 4.

## COMPARISON OF DESIGN AND OPERATIONAL MISSION PROFILE FATIGUE LOAD SPECTRA

The fatigue load spectra for the main rotor and tail rotors fall into three data categories:

1. Applicable to design spectrum only

2. Applicable to operational spectrum only

3. Applicable to both spectra, load same or different

Consult the percentage of occurrence columns of the detailed conditions listing (Table II) to ascertain the appropriate category. Main rotor and tail rotor loads for individual detailed conditions that show a percentage of occurrence for both spectra can be compared directly in Table VI. Inasmuch as the tail rotor fatigue data were not added to the histogram (Figure 4) as previously discussed, a histogram comparison of tail rotor data for the design and operational mission profiles is not directly available. However, by using the tail rotor loads (Table VI) along with the appropriate percentage of occurrence for each mission profile (Table II), good fatigue load spectra correlation is shown between the design and operational data.

## TABLE IX. OPERATIONAL MISSION PROFILE — $C_T/\sigma$ AND μ
### (See Notes for Values Used in Calculations)

| Detailed Condition Number | $C_T/\sigma$ | μ | (1) ± inch-pounds | Detailed Condition Number | $C_T/\sigma$ | μ | (1) ± inch-pounds |
|---|---|---|---|---|---|---|---|
| II-c 1 | 0.146 | 0.097 | 250 | II-e 10 | 0.174 | 0.299 | 2210 |
| II-c 2 | 0.155 | 0.100 | 250 | II-j 1 | 0.108 | 0.091 | 250 |
| II-c 3 | 0.167 | 0.097 | 250 | II-j 2 | 0.135 | 0.102 | 250 |
| II-c 4 | 0.177 | 0.100 | 250 | II-j 3 | 0.148 | 0.091 | 250 |
| II-c 5 | 0.146 | 0.104 | 300 | II-j 4 | 0.185 | 0.102 | 250 |
| II-c 6 | 0.155 | 0.200 | 360 | II-j 5 | 0.108 | 0.182 | 290 |
| II-c 7 | 0.167 | 0.194 | 620 | II-j 6 | 0.135 | 0.204 | 310 |
| II-c 8 | 0.177 | 0.200 | 990 | II-j 7 | 0.148 | 0.182 | 290 |
| II-c 9 | 0.122 | 0.291 | 610 | II-j 8 | 0.185 | 0.204 | 1280 |
| II-c 10 | 0.129 | 0.299 | 940 | II-j 9 | 0.108 | 0.274 | 410 |
| II-c 11 | 0.146 | 0.291 | 1320 | II-j 10 | 0.135 | 0.306 | 1200 |
| II-c 12 | 0.155 | 0.299 | 1680 | II-j 11 | 0.148 | 0.274 | 1160 |
| II-c 13 | 0.167 | 0.291 | 1910 | II-j 12 | 0.162* | 0.306 | 1970* |
| II-c 14 | 0.177 | 0.299 | 2310 | II-j 13 thru II-j 24, same as 1 thru 12 | | | |
| II-c 15 thru II-c 28, same as 1 thru 14 | | | | II-1 1 | 0.108 | 0.152 | 270 |
| II-e 1 | 0.146 | 0.162 | 280 | II-1 2 | 0.135 | 0.170 | 280 |
| II-e 2 | 0.155 | 0.166 | 280 | II-1 3 | 0.145 | 0.152 | 270 |
| II-e 3 | 0.164 | 0.162 | 280 | II-1 4 | 0.181 | 0.170 | 710 |
| II-e 4 | 0.174 | 0.166 | 440 | II-1 5 | 0.108 | 0.274 | 410 |
| II-e 5 | 0.122 | 0.291 | 610 | II-1 6 | 0.135 | 0.306 | 1200 |
| II-e 6 | 0.129 | 0.299 | 940 | II-1 7 | 0.145 | 0.274 | 1070 |
| II-e 7 | 0.146 | 0.291 | 1320 | II-1 8 | 0.148 | 0.306 | 1570* |
| II-e 8 | 0.155 | 0.299 | 1680 | IV-c 19 | 0.109 | 0.262 | 380 |
| II-e 9 | 0.164 | 0.291 | 1820 | IV-c 20 | 0.115 | 0.270 | 400 |

NOTES: (1) Cyclic M/R 15% Flapwise Bending from Figure 5
(2) Power on rpm: max 484; min 470
   Autorotation: max 514; min 400 (460 used as minimum, as shown in Reference 1, Figure 6.b.
(3) Hd = 2000 ft, except in Table IV-c 19 and 20 where Hd = 3000 ft
(4) Load factors used are as shown in Table V
(5) Gross weight = 2400 lb (OH-6A normal gross weight) except Table II-e 3, 4, 9, 10 and Table II-1 3, 4, 7, 8 where gross weight = 2200 lb
(6) $V_{NE}$ = 124 knots calibrated airspeed
(7) $A_b$ = main rotor blade area = 29.625 ft$^2$

*Ratio to load factor = 1.8 g per Reference 1, Figure 14 which shows original calculated $C_T/\sigma$ and μ combination do not occur in actual operation.

Figure 3 shows a main rotor fatigue load histogram comparison of mission profiles directly. For purposes of this discussion, three ranges of main rotor load were considered: (1) less than 800 inch-pounds, (2) 800 to 1600 inch-pounds, and (3) greater than 1600 inch-pounds.

The majority of the time for both mission profiles occurs in the lowest load range being considered. This is particularly important inasmuch as 800 inch-pounds represents the approximate main rotor blade endurance limit*. The design spectrum data are shifted to a slightly higher load level than the operational data. However, loads in this range do not affect component service life.

The comparison of design and operational loads in the second range (800 to 1600 inch-pounds) is characterized by small offsetting percentage of occurrence differences. One major exception, however, is shown for the load between 1200 and 1300 inch-pounds. The design spectrum percentage of occurrence exceeds the corresponding operational value by several percent.

While the peak load that occurs in the final or highest load range originated in the design mission profile, the operational mission profile produced more high loads overall.

With the exception of the differences previously noted, the two mission profile fatigue load histograms compare quite favorably. The comparison is so close that the mission profile impact on main rotor blade service life cannot be seen from Figure 3, but will have to await more extensive analysis in the discussions that follow.

## DETERMINATION OF FATIGUE DAMAGE RATES AND FATIGUE LIVES

During the design effort the damaging flight conditions of the design spectrum were tabulated together with their respective measured peak alternating flap bending moments, percentage of occurrences and computed damage rates (Reference 12).

The ground-air-ground (GAG) condition included in the spectrum in Reference 12 was omitted from the design spectrum, but the damaging cycles previously taken from design flight condition data to form a part of that ground-air-ground condition, were replaced in their respective individual conditions, and the revised damage rates for the concerned conditions

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*Fatigue damage occurs only at moments greater than this magnitude.

recalculated. This resulted in a total damage rate $(\Sigma \frac{n}{N})$ of 0.2950 per 1000 hours for the flight conditions of the design spectrum.

The time histories of loads obtained during flight for most of the damaging conditions in the design spectrum had been cycle-counted, with the damage rates for a few conditions calculated on a one-per-revolution basis (Reference 12). However, developed data for the operational mission profile, Table II and VI, present only peak alternating moments and percentages of occurrence. After preliminary investigation, it was concluded that damage rates calculated on a one-cycle-per-revolution basis should be used to compare the design spectrum with the operational mission profiles, and the following procedure was utilized.

Based on a main rotor speed of 475 rpm and the given percentages of occurrences, the effective number of cycles per 1000 hours, $n_e$, was calculated for each damaging flight condition of the design spectrum; then, using the damage rate, $\frac{n}{N}$, for each condition as established in the life calculation and $n_e$, an effective allowable number of cycles, $N_e$, was derived. From the established S-N curve for 15 percent blade radius, the corresponding effective alternating moment, $M_e$, was then obtained for each value of $N_e$. Thus, for each flight condition, the damage rate due to this effective alternating moment, $M_e$, at one-per-revolution was identical to the damage rate obtained for that condition by cycle counting.

A plot of $M_e$ versus $M_{peak}$ was then made for each previously cycle-counted condition of the design spectrum, Figure 6. From these points, a curve was drawn such that the total damage rate for all damaging conditions of the design spectrum (derived by the use of this curve and the given percentage of occurrences at one-per-revolution and 475 rpm) was identical to the total damage rate for the same conditions as established in the original life calculation including cycle-counting.

Using the curve of Figure 6 together with the previously established safe allowable S-N curve for 15 percent blade radius, the damage rates for the damaging conditions of the operational mission profile (TR 71-60) were calculated based on the given peak alternating moments and percentage of occurrences. These data are presented in Table X.

The total damage rate so obtained per 1000 hours for the flight conditions of the operational mission profile was 0.3144 or slightly greater than the value of 0.2950 obtained for the flight conditions of the design spectrum.
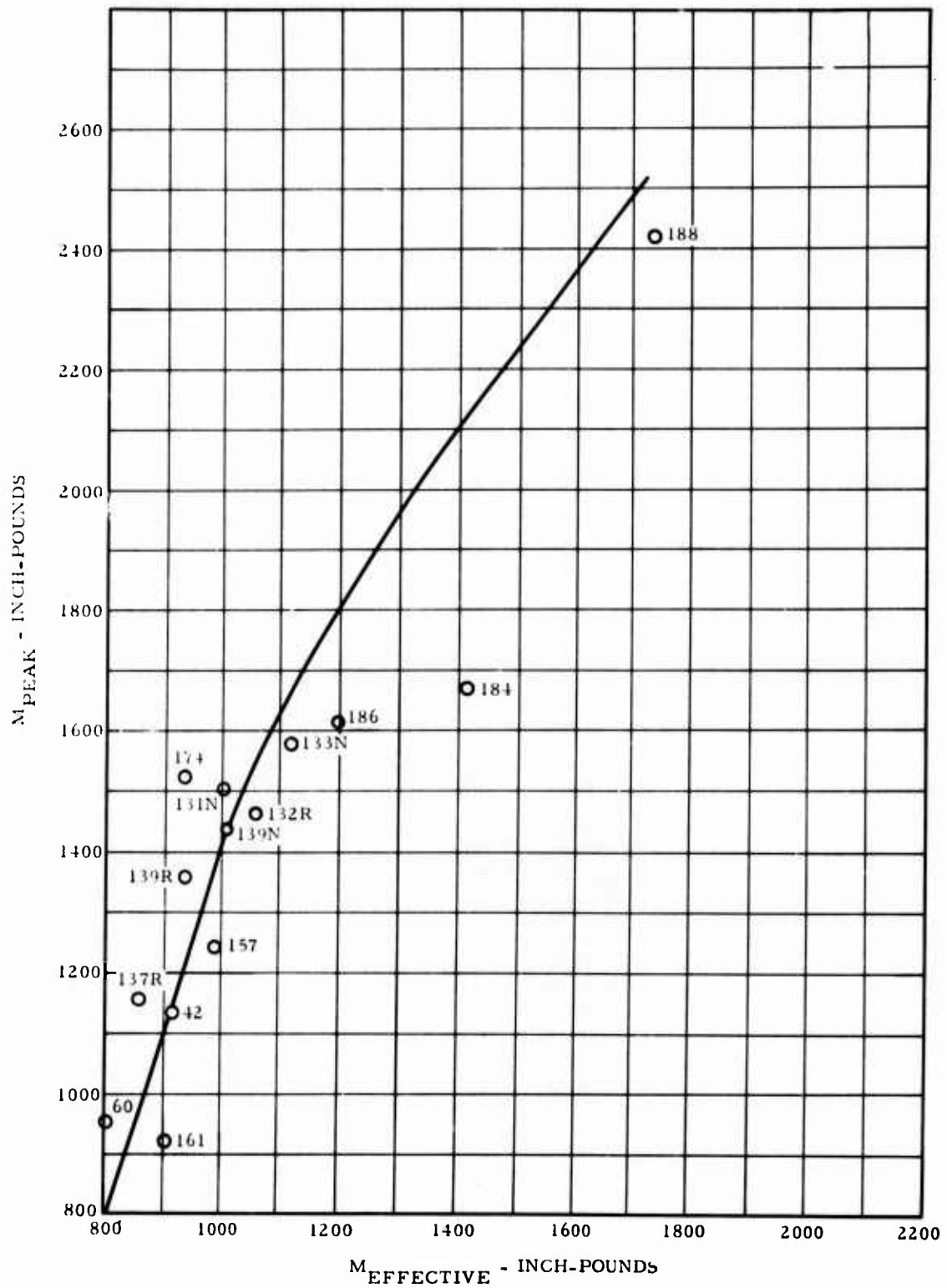
30

Figure 6.   Peak Alternating Moment Versus Effective Alternating Moment.

| | | | | | | | |
|---|---|---|---|---|---|---|---|
| TABLE X. OPERATIONAL MISSION PROFILE DAMAGE RATE |

| Mission Segment | Condition Number | n (475 rpm) | $M_{PEAK}$ | $M_e$ | Percentage of Occurrence | N | $\frac{n}{N}$ |
|---|---|---|---|---|---|---|---|
| II  Maneuver | c 8 | 0.00305 | 990 | 862 | 0.0107 | 18.25 | 0.000167 |
| | c 10 | 0.09776 | 940 | 846 | 0.3430 | 23.20 | 0.004213 |
| | c 11 | 0.00182 | 1320 | 971 | 0.0004 | 5.55 | 0.000329 |
| | c 12 | 0.00182 | 1680 | 1125 | 0.0064 | 2.05 | 0.000890 |
| | c 13 | 0.00305 | 1910 | 1265 | 0.0107 | 1.05 | 0.002904 |
| | c 14 | 0.00305 | 2310 | 1561 | 0.0107 | 0.31 | 0.009836 |
| | c 22 | 0.00305 | 990 | 862 | 0.0107 | 18.25 | 0.000167 |
| | c 24 | 0.09776 | 940 | 846 | 0.3430 | 23.20 | 0.004214 |
| | c 25 | 0.00182 | 1320 | 971 | 0.0064 | 5.55 | 0.000329 |
| | c 26 | 0.00182 | 1680 | 1125 | 0.0064 | 2.05 | 0.000890 |
| | c 27 | 0.00305 | 1910 | 1265 | 0.0107 | 1.05 | 0.002904 |
| | c 28 | 0.00305 | 1310 | 1561 | 0.0107 | 0.31 | 0.009837 |
| | e 6 | 0.03175 | 940 | 846 | 0.1114 | 23.20 | 0.001308 |
| | e 7 | 0.00123 | 1320 | 971 | 0.0043 | 5.55 | 0.000221 |
| | e 8 | 0.00123 | 1680 | 1125 | 0.0043 | 2.05 | 0.000598 |
| | e 9 | 0.00123 | 1820 | 1205 | 0.0043 | 1.27 | 0.000961 |
| | e 10 | 0.00123 | 2210 | 1483 | 0.0043 | 0.42 | 0.002918 |
| | f 1 | 0.02671 | 1463 | 1073 | 0.0940 | 3.80 | 0.007029 |
| | f 4 | 0.7410 | 1580 | 1075 | 0.2210 | 2.75 | 0.026945 |
| | f 6 | 0.2671 | 998 | 865 | 0.0940 | 17.50 | 0.001525 |
| | f 7 | 0.6298 | 832 | 810 | 0.2210 | 56.00 | 0.001125 |
| | f 8 | 0.06298 | 1503 | 1041 | 0.2210 | 3.38 | 0.010745 |
| | f 9 | 0.02671 | 862 | 820 | 0.0940 | 40.00 | 0.000668 |
| | f 11 | 0.06298 | 871 | 823 | 0.2210 | 16.00 | 0.003936 |
| | f 12 | 0.06298 | 1083 | 893 | 0.2210 | 12.00 | 0.005248 |
| | g 1 | 0.02964 | 913 | 837 | 0.1040 | 27.00 | 0.001098 |
| | j 8 | 0.00012 | 1280 | 958 | 0.0007 | 6.20 | 0.000032 |
| | j 10 | 0.00664 | 1200 | 931 | 0.0233 | 7.90 | 0.000341 |
| | j 11 | 0.00105 | 1160 | 918 | 0.0064 | 9.10 | 0.000116 |
| | j 12 | 0.00012 | 1970 | 1305 | 0.0007 | 0.88 | 0.000227 |
| | j 20 | 0.00012 | 1280 | 958 | 0.0007 | 6.20 | 0.000032 |
| | j 22 | 0.00664 | 1200 | 931 | 0.0233 | 7.90 | 0.000841 |
| | j 23 | 0.00105 | 1160 | 918 | 0.0064 | 9.10 | 0.000116 |
| | j 24 | 0.00012 | 1970 | 1305 | 0.0007 | 0.88 | 0.000227 |
| | k 2 | 0.00536 | 924 | 841 | 0.0188 | 25.00 | 0.000214 |
| | k 4 | 0.01268 | 978 | 858 | 0.0445 | 19.00 | 0.000667 |
| | l 6 | 0.00658 | 1200 | 931 | 0.0231 | 7.90 | 0.000833 |
| | l 7 | 0.00219 | 1070 | 889 | 0.0077 | 12.80 | 0.000171 |
| | l 8 | 0.00026 | 1570 | 1072 | 0.0009 | 2.78 | 0.000092 |
| III  Descent | a 5 | 0.04845 | 1689 | 1316 | 0.1700 | 0.84 | 0.058020 |
| | a 6 | 0.04845 | 1689 | 1316 | 0.1700 | 0.84 | 0.058020 |
| | b 1 | 0.48735 | 952 | 850 | 1.1700 | 21.50 | 0.022670 |
| | b 3 | 0.24367 | 1179 | 924 | 0.8550 | 8.50 | 0.028670 |
| | b 4 | 0.12255 | 1179 | 924 | 0.4300 | 8.50 | 0.014420 |
| | c 1 | 0.97755 | 860 | 820 | 3.4300 | 40.00 | 0.024440 |
| IV  Steady State | c 36 | 0.02987 | 835 | 811 | 0.1048 | 54.00 | 0.000553 |
| | c 37 | 0.01824 | 810 | 803 | 0.0640 | 80.00 | 0.000228 |
| | c 38 | 0.01824 | 1009 | 869 | 0.0640 | 16.60 | 0.001099 |
| | c 40 | 0.00329 | 1280 | 958 | 0.0119 | 6.20 | 0.000547 |
| | c 41 | 0.00117 | 956 | 851 | 0.0041 | 21.10 | 0.000055 |
| | c 42 | 0.00111 | 1321 | 971 | 0.0039 | 5.55 | 0.000200 |

$$\Sigma \frac{n}{N} = 0.314396$$

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Table I presents the basic flight conditions for the operational mission profile derived from TR 71-60, the AR-56 mission profile, and the AMCP 706-203 mission profile, together with their respective percentage of occurrences. Table XI presents the predicted damage rate due to these basic flight conditions for each of these three mission profiles together with the damage rate for the design spectrum. The damage for the basic conditions in the TR 71-60 operational mission profile was obtained by the summation of the appropriate detailed conditions in Table X. The predicted damage due to the basic conditions in the AR-56 and the AMCP 706-203 profiles was obtained by factoring the damage for these basic conditions due to the TR 71-60 profile by the ratio of the total percentage of occurrences for the basic condition in the respective profiles.

From the cumulative damage rates for the three mission profiles derived from Table XI, comparative service lives were calculated and presented as a bar chart on Figure 7. These lives do not reflect a ground-air-ground condition. The original design spectrum service life is 2520 hours. The calculated life for the operational spectrum, 2390 hours, is slightly lower.

The service lives presented were established in accordance with the formulae used by the Federal Aviation Agency (CAM 6, Reference 4) as follows:

1.  Calculated life, $L_c$,        $\leq 3350$ hours

    Service life, $L_s$,        $= 0.75 L_c$

2.  Calculated life, $L_c$,        $\geq 3350$

    Service life, $L_s$,        $= 0.375 L_c + 1250$ hours

The loads data and the corresponding damage rates for the design spectrum and the operational profile were studied with a view to establishing some definable relationships between peak alternating moment, percentage of occurrence and damage rate. The following paragraphs summarize the results of this study and the subsequent conclusions.

Figure 8 presents the comparison of cumulative damage rates for the design spectrum and operational mission profile, and Figure 9 presents the comparison of cumulative percentage of occurrence for the same two spectra. The data for these two histograms were obtained from Tables II and X.

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TABLE XI. DAMAGE RATE COMPARISON

Mission Profile

| Mission Segment | Design | | Operational | | AR-56 Utility | | AMCP 706-203 | |
|---|---|---|---|---|---|---|---|---|
| | Percentage of Occurrence | Damage Rate | Percentage of Occurrence | Damage Rate | Percentage of Occurrence | Damage Rate | Percentage of Occurrence | Damage Rate |
| **II Maneuver** | | | | | | | | |
| c | 6.00 | 0.0171 | 8.42 | 0.03668 | 4.20 | 0.01830 | 2.85 | 0.01241 |
| e | 1.00 | 0.0766 | 1.40 | 0.00607 | 0.18 | 0.00078 | 0.89 | 0.00386 |
| f | 1.50 | 0.0812 | 2.21 | 0.05722 | 0.59 | 0.15280 | 0.06 | 0.00155 |
| g | 0.10 | 0.0010 | 0.15 | 0.00110 | 0 | 0 | 0 | 0 |
| j | 2.00 | 0.0119 | 2.80 | 0.00243 | 0.36 | 0.00031 | 0 | 0 |
| k | 1.50 | 0.0439 | 2.21 | 0.00088 | 0.26 | 0.00010 | 0 | 0 |
| l | 1.00 | 0 | 1.40 | 0.00110 | 0.03 | 0.00002 | 0 | 0 |
| **III Descent** | | | | | | | | |
| a | 2.00 | 0.0181 | 3.43 | 0.11604 | 4.89 | 0.16543 | 4.84 | 0.16402 |
| b | 3.00 | 0.0265 | 5.14 | 0.06576 | 3.06 | 0.03915 | 2.63 | 0.03365 |
| c | 2.00 | 0.0143 | 3.43 | 0.02444 | 0.26 | 0.00185 | 0.01 | 0.00007 |
| **IV Steady State** | | | | | | | | |
| c | 65.60 | 0.0044 | 48.79 | 0.00268 | 67.33 | 0.00370 | 70.57 | 0.00386 |
| $\Sigma \frac{n}{N}$ | | 0.2950 | | 0.31440 | | 0.24492 | | 0.21942 |

$L_c = 3390$    $L_c = 3181$    $L_c = 4083$    $L_c = 4557$

$L_s = 2520$    $L_s = 2390$    $L_s = 2780$    $L_s = 2960$

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MANEUVER SPECTRUM

Figure 7. Service Life Comparison.

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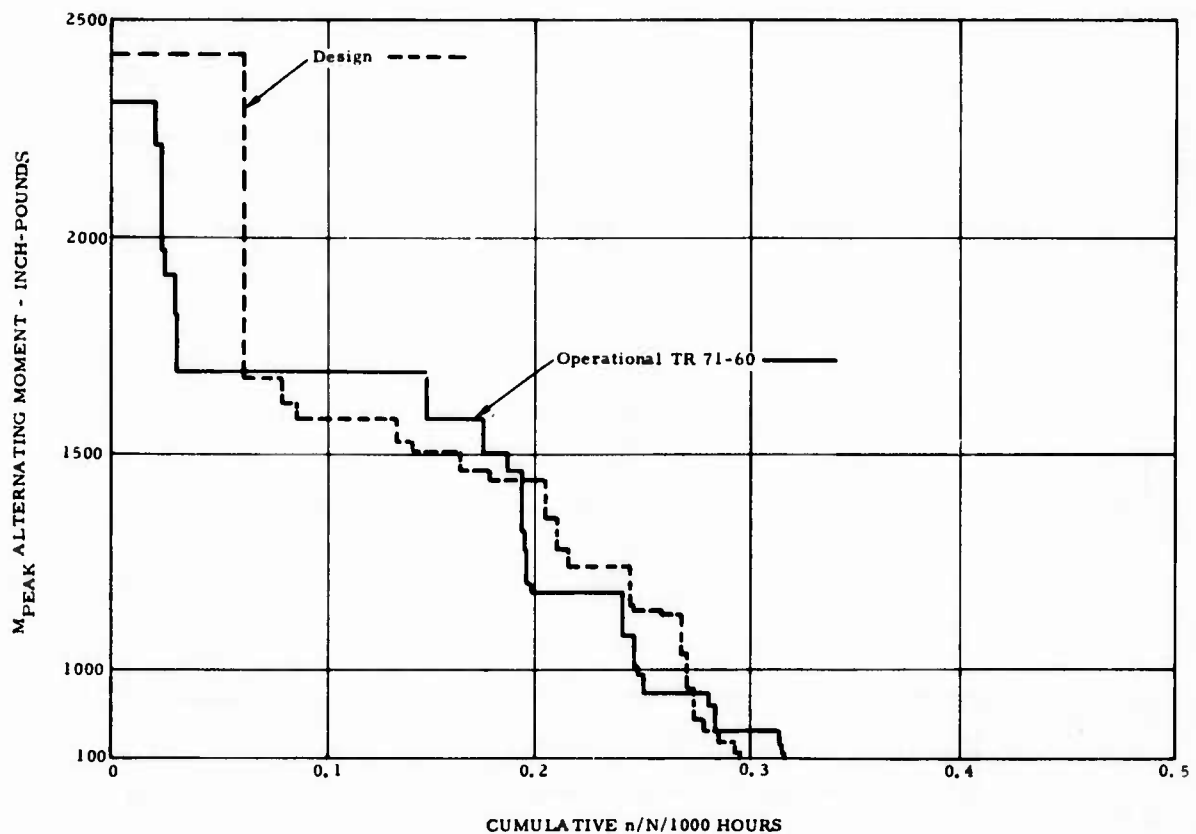
Figure 8. Comparison of Cumulative Damage Rates.

Superimposed on Figure 9 is a single representative curve for peak moment versus cumulative percentage of occurrence, which was transposed from the straight line curve of Figure 10. These curves were derived in the following manner:

An approximate mean smooth curve of the two stepped curves of Figure 9 was first drawn. Study showed that this mean curve could be closely represented by a straight line curve on a Log-Log scale. The two points chosen to define the straight line curve were: $M_p$ = 2250 inch-pounds at 0.1 cumulative percentage of occurrence, and $M_p$ = 800 inch-pounds (the endurance limit of the blade) at 10 cumulative percentage of occurrence

36
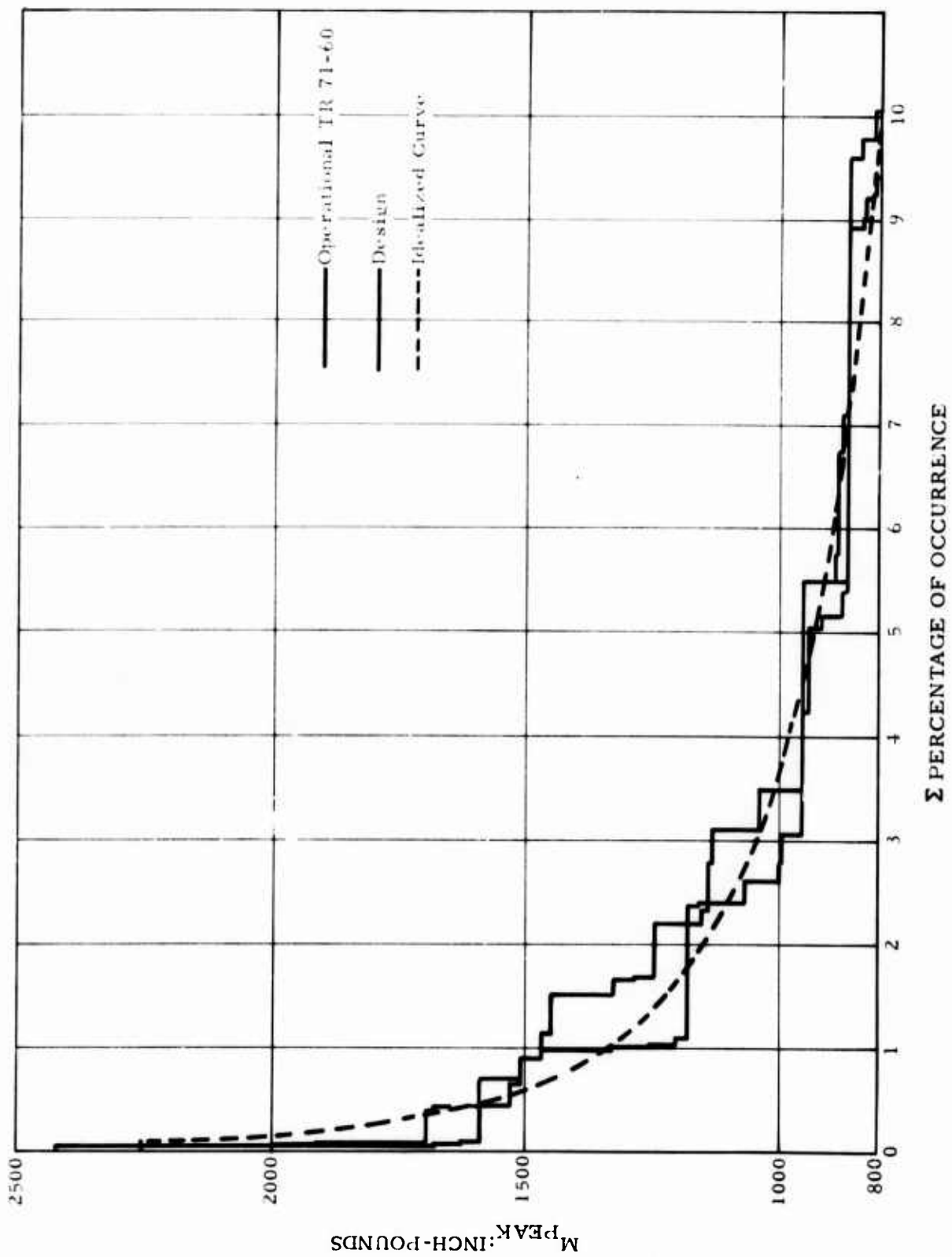
Figure 9.   Comparison of Cumulative Percentage of Occurrence
Versus Peak Moment.

37
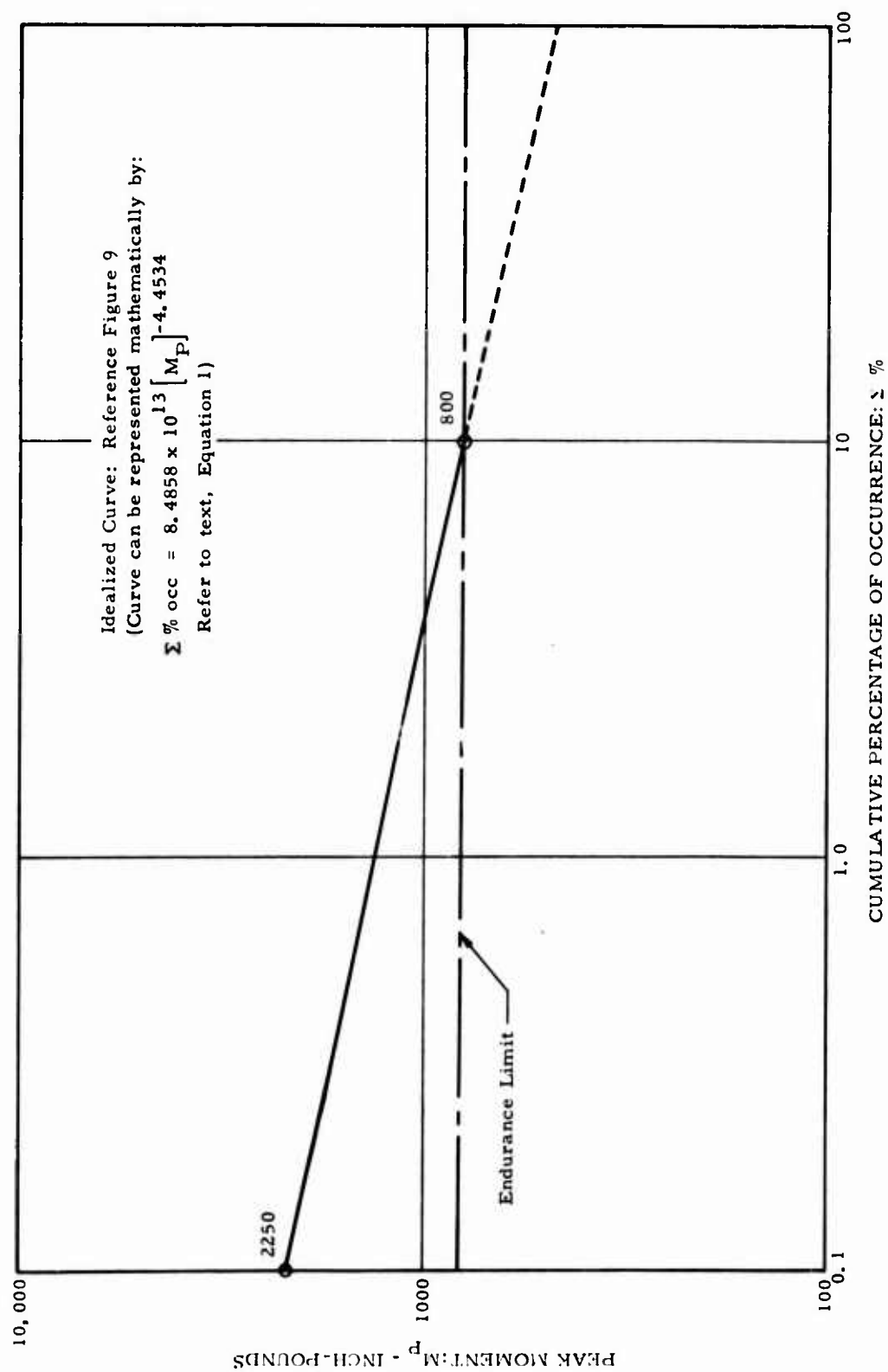
Figure 10. Cumulative Percentage of Occurrence Versus Peak Moment.

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(which includes all the damaging conditions). This mathematically representative curve of peak moment versus cumulative percentage of occurrence of Figure 10 was then superimposed on Figure 9.

$$\text{Cumulative Percentage Occurrence, } \Sigma\%_{\text{occur}} = 8.4858 \times 10^{13} \left[ M_P \right]^{-4.4534}$$
(1)

The increment of percentage occurrence for an increment of peak moment is a function of the rate of change of cumulative percentage of occurrence with respect to peak moment at the chosen level of peak moment. The total cumulative percentage of occurrence during which damage occurs between the maximum peak moment and the endurance limit of 800 inch-pounds is 10 percent. (Reference Figure 9.)

Thus, differentiating equation 1 with respect to 100 inch-pound increments of peak moment:

$$\text{Percentage of Occurrence} = -3.7721 \times 10^{16} \left[ M_P \right]^{-5.4534}$$
(2)

Equation 2 defines the curve on Figure 11.

Figure 6 converts peak moment to the required effective moment as defined previously. The previously established S-N curve for the 15-percent station of the OH-6A main rotor blade, Figure 12, was transposed using Figure 6 to present peak moment versus allowable number of cycles, N, as presented on Figure 13.

For unit percentage of occurrence and a rotor speed of one revolution per minute, the number of cycles of moment per 1000 hours, n, = 600. Thus, Figure 13 can be replotted to present peak moment versus damage rate per percentage occurrence per rpm per 1000 hours, as in Figure 14. This curve can be defined by the equation:

$$\text{Log}_{10}\left[ \frac{n}{N} \right]_u = 0.3979 + 0.001236 \left[ M_P \right]$$
(3)

39
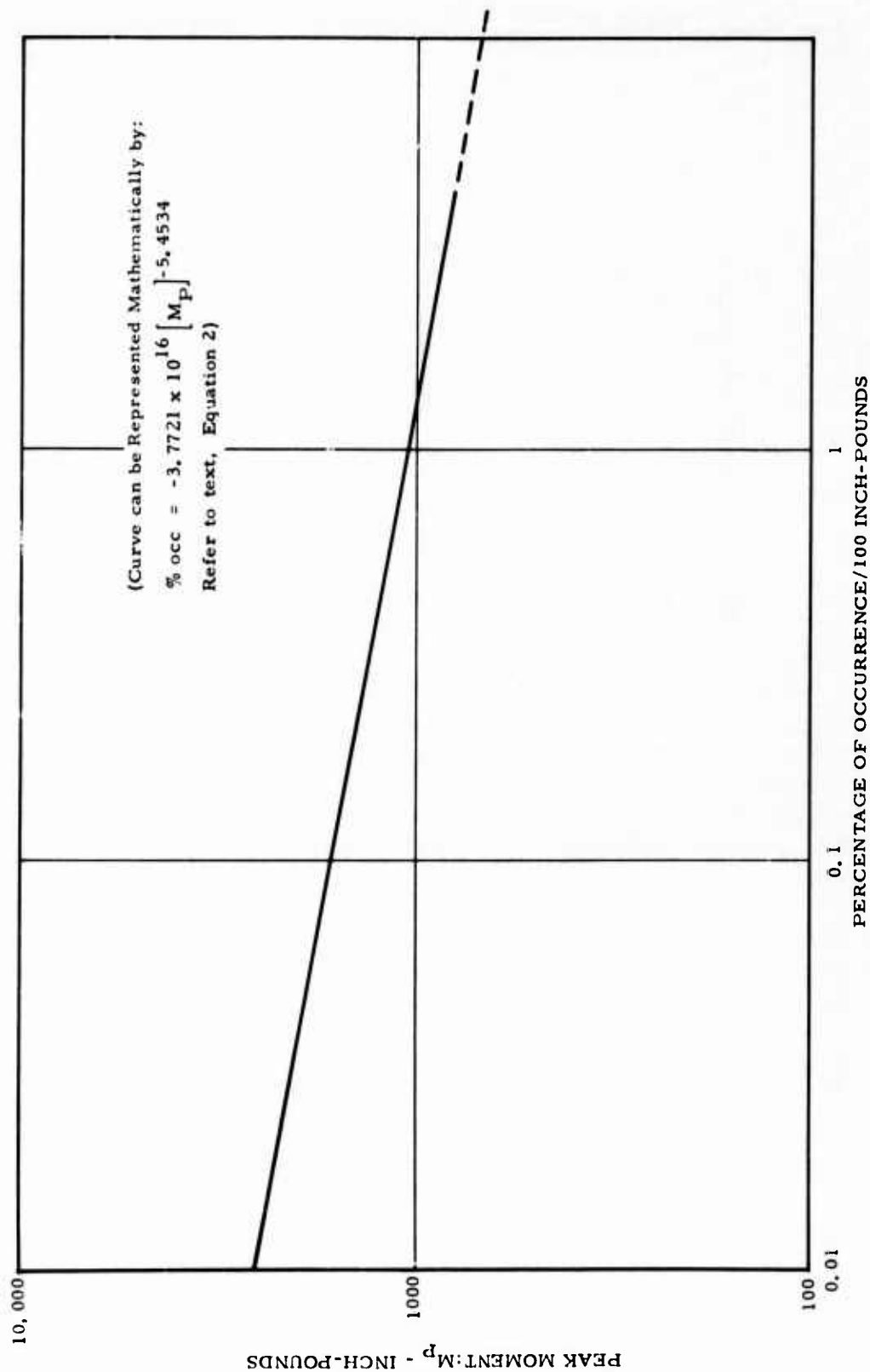
Figure 11. Percentage of Occurrence Per 100 Inch-Pounds Increment of Moment.

40
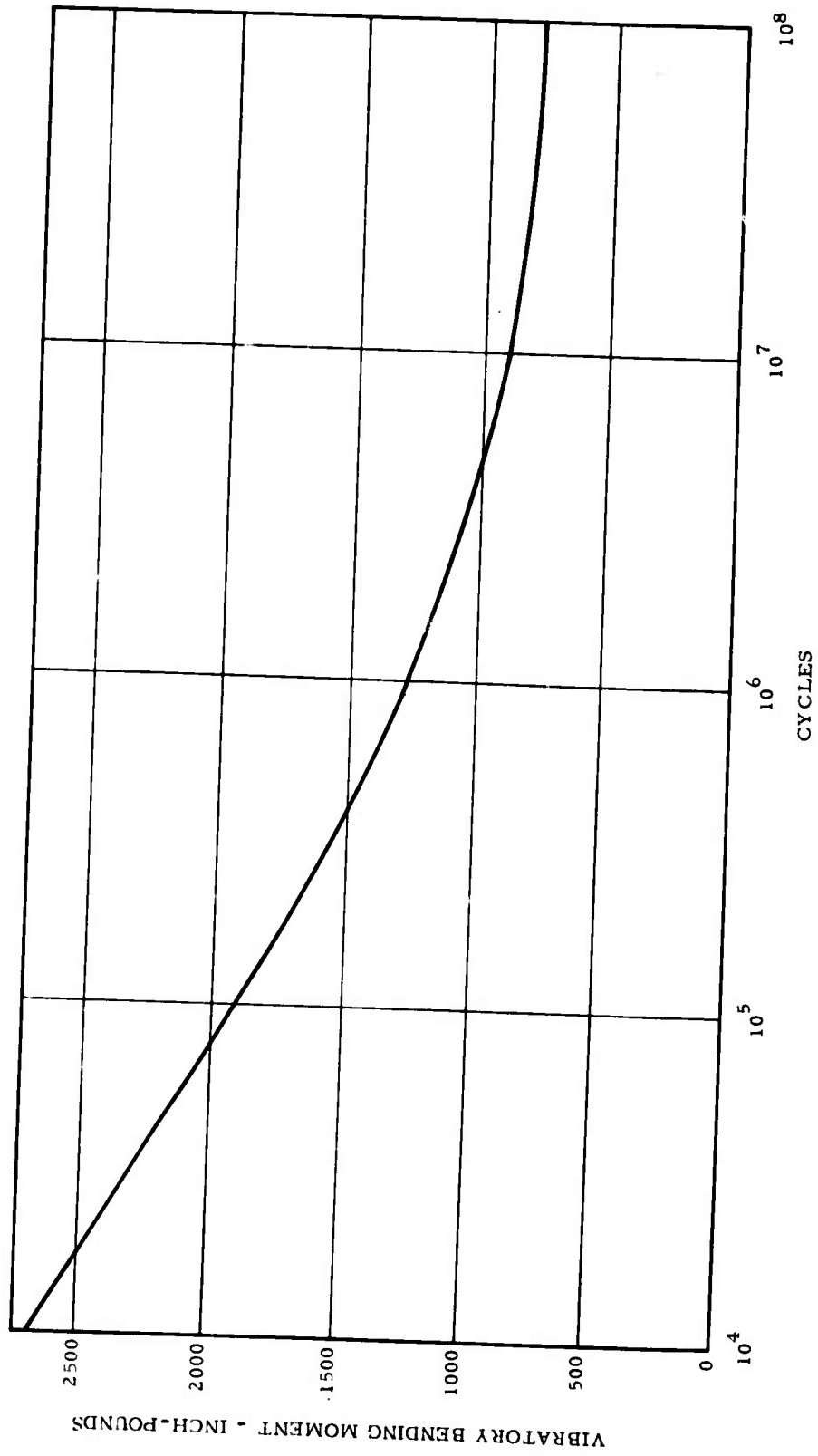
Figure 12.   S-N Curve for 369 (OH-6A) Main Rotor Blade Fatigue Specimens.

41
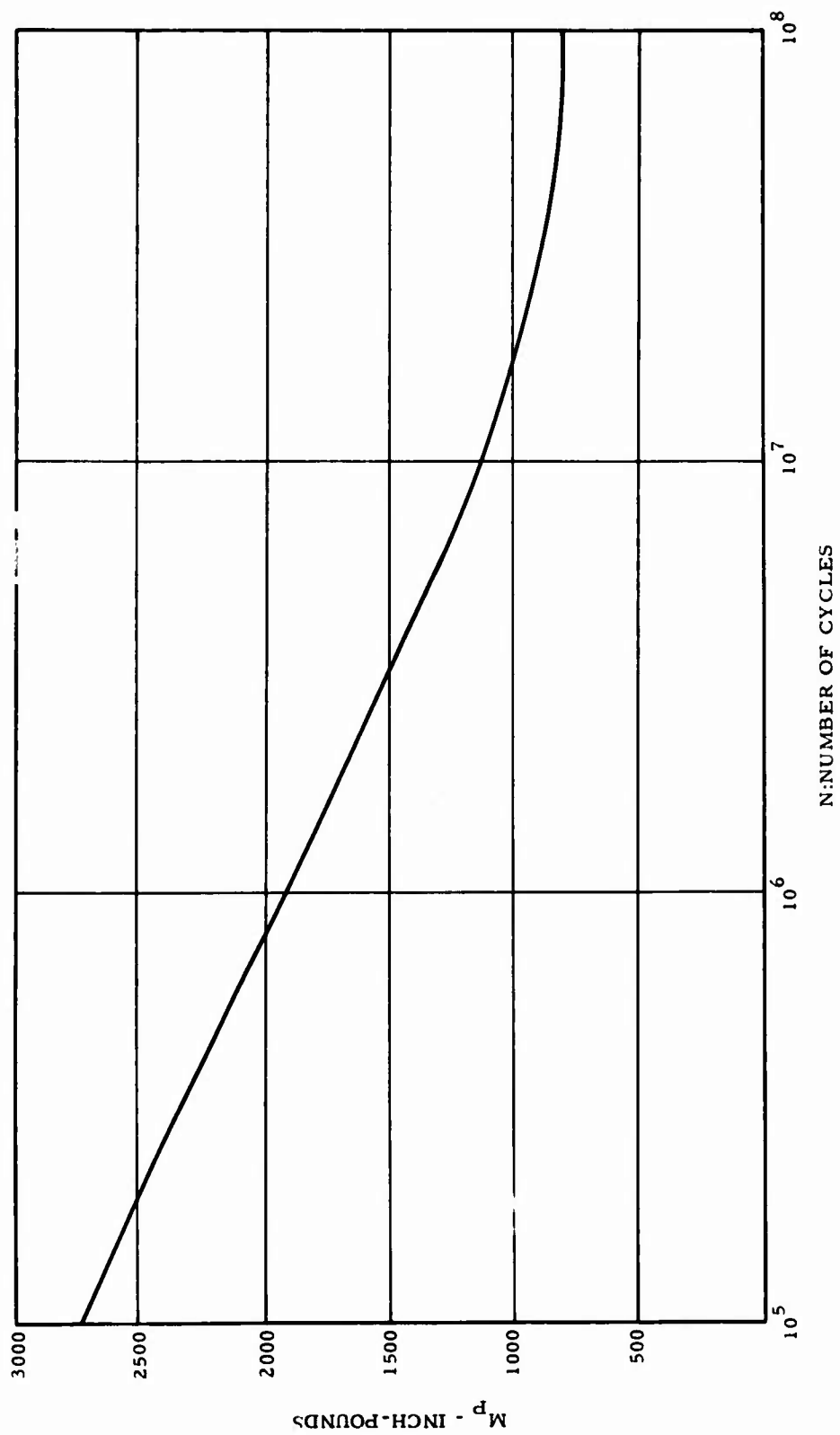
Figure 13. Peak Moment Versus Allowable Cycles.

42

(Curve can be Mathematically Represented by:

$$LOG_{10}\left[\frac{n}{N}\right]_u = 0.001236\left[M_P\right] + 0.3979$$

Refer to Text, Equation 3)

$\left[(n/N)/\text{PERCENTAGE OF OCCURRENCE/RPM/1000 HOURS}\right] \times 10^6$
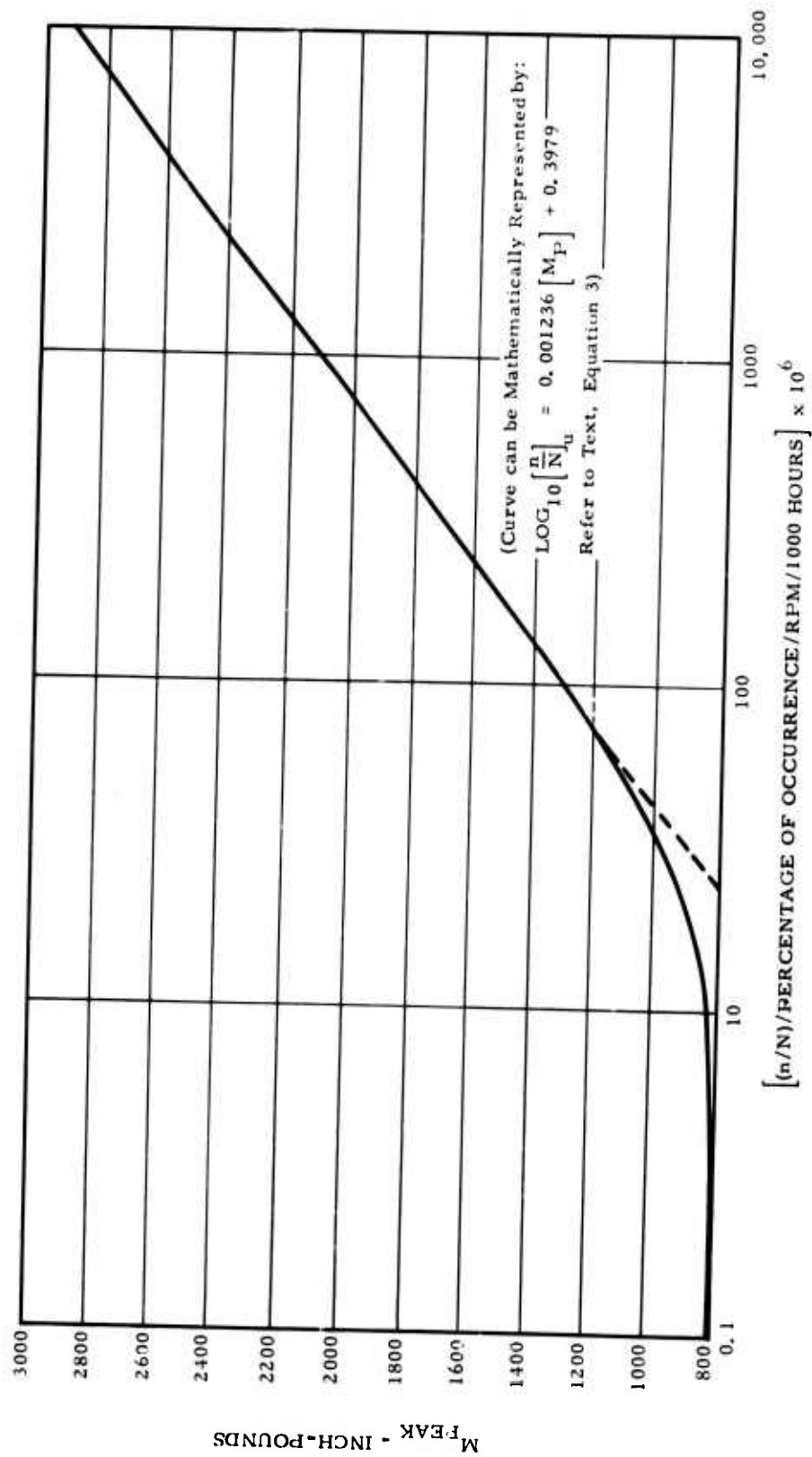
$M_{PEAK}$ - INCH-POUNDS

Figure 14. Peak Moment Versus Damage Rate/Percentage of Occurrence/RPM/1000 Hours.

43

where

$$\left[\frac{n}{N}\right]_u = \frac{n}{N}/\% \text{ occurrence/rpm/1000 hours}$$

For a maximum peak alternating moment of 2300 inch-pounds as defined by the operational mission profile, together with the established endurance limit of 800 inch-pounds, the total damage rate per 1000 hours computed by the use of equations 2 and 3 is 0.3099 compared to 0.3144 given by the detailed operational loads spectrum in Table X, which is in remarkably close agreement.

A comparison of the results given in the above paragraph indicates that for the OH-6A main rotor blade, relationships between peak alternating moments, percentages of occurrences and damage rates can be defined from the given load spectrum and established S-N curve, and that these relationships could be utilized in the design stages of a similar blade in order to achieve the desired service life for a defined operational spectrum.

## TASK III - ANALYSIS OF HISTORICAL CHANGES IN FATIGUE LIVES AND CONFIGURATIONS

### INTRODUCTION

Historical data concerning changes in component lives and configurations from the time of original engineering development to the present are discussed. The changes are analyzed for cause-effect relationships with actual changes in mission profiles as compared to developmental flight spectra. The components studied are main and tail rotor blades and drive system.

### MAIN ROTOR BLADE

The helicopter main rotor blade is the most critical structural component of a helicopter. The failure of other primary structural components may cause emergency landings; the loss of a helicopter main rotor blade is catastrophic. A large amount of time and effort is expended in the design of main rotor blades, including prediction of loads, frequencies, static and fatigue strengths, selection of materials, and methods of manufacturing and processing.

The service life of a main rotor blade more closely reflects the effects of various parameters, such as gross weight, rpm, maneuvers, speed, etc, than any other single structural component. The only major load parameter not reflected in the main rotor blade service life is changes in engine power. The drive system transmissions and shafts are the structural components affected by changes in engine power spectra. The main rotor blade is, therefore, selected as the structural component to use to relate the mission profiles to the original design spectrum. The results of this comparison are presented in Task II of this report.

#### History

The selection of the main rotor blade for the prototype OH-6 was based on the experience, knowledge and testing of the Model 269A (TH-55A) helicopter main rotor blade. The blade cross section was essentially the same as the Model 269 using an aluminum alloy leading edge spar and a 0.025-inch-thick aluminum skin. There were several structural modifications made

primarily to increase the blade fatigue strength or reduce cyclic loads, such as:

1. The root fitting was bonded to the blade skin doubler with bolts retained to provide a redundant or fail-safe attachment and to prevent prying from occurring on the adhesive bond line.

2. The trailing edge was bonded instead of a riveted connection as on the Model 269 blade.

3. A doubler was added to the root end of the blade to improve the fatigue strength at the root end fitting attachment.

4. A channel was added internally 4.4 inches aft of the leading edge to maintain the blade airfoil shape, thereby improving performance and reducing cyclic loads.

5. A leading edge weight was added and the tip weight design was changed to improve blade mass balance.

Figure 15 shows the cross sections of the OH-6 and Model 269 main rotor blades with the main design differences noted.

The prototype OH-6 main rotor blades were fatigue tested. The results of the tests showed an improvement in fatigue strength compared to the Model 269A main rotor blades although the mode of failure was the same. The critical section was the same, the basic blade section at the root fitting outboard bolt hole.

Detail changes were made in the construction of the production OH-6A main rotor blades to provide additional improvement in fatigue strength, mass balance and stiffening to maintain airfoil shape. The changes were:

1. The root fitting changed from a stepped machine fitting to a tapered forged fitting.

2. The shape of the root end doubler revised.

3. The leading edge spar tapered on the trailing edge.

4. The brass leading edge balance weight increased in size and changed in shape.

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Alum Spar

0.025 In. Alum Skin
Riveted Trailing Edge

Model 269A Blade - Dwg 269B1145

Brass Leading Edge Weight

Channel
Bonded Trailing Edge

Model OH-6 Blade - Dwg 369-1140

Changed Shape of Leading Edge Spar and Balance Weight

Model OH-6A Blade - Dwg 369A1100
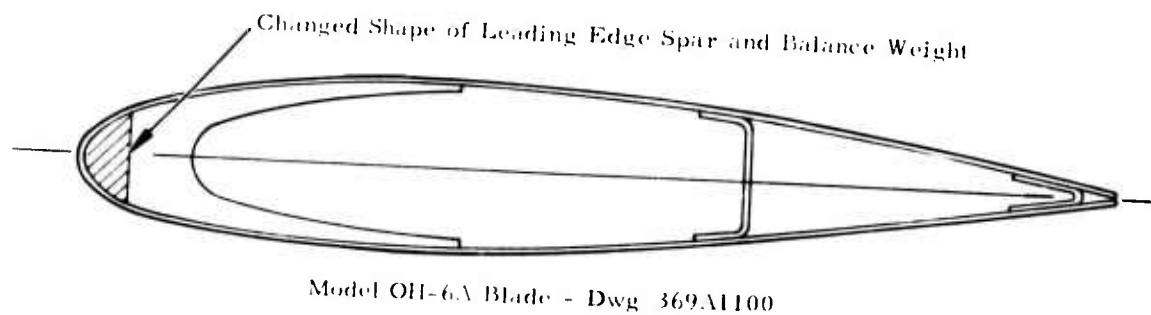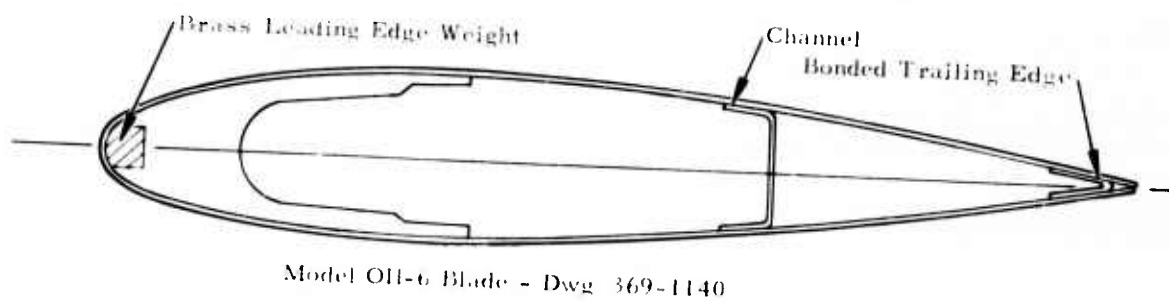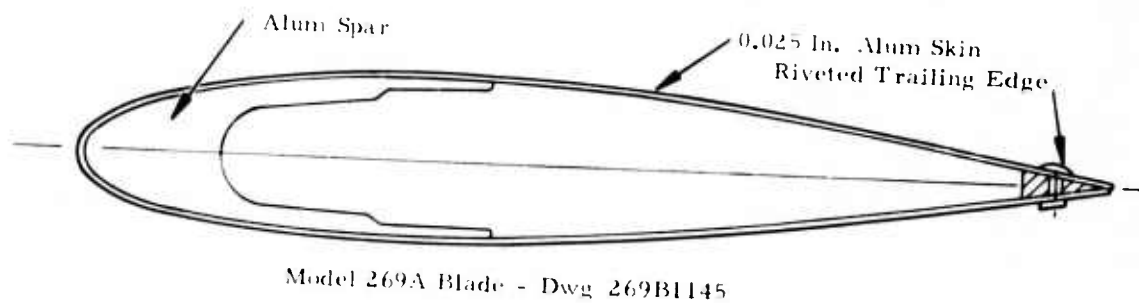
Figure 15.   Main Rotor Blade Cross Section.

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5. The tip balance weight installation improved.

6. Additional chordwise ribs added at the blade tip.

Figure 16 shows the configuration changes.

The configuration changes resulted in higher fatigue strength main rotor blades and lower cyclic loads for similar flight conditions. The mode of failure of the main rotor blade shifted from the basic blade section at the outboard bolt hole to the lugs of the root fittings. To obtain additional fatigue strength in the root area of the main rotor blade it will be necessary to have the root fittings manufactured from a higher fatigue strength material, or to use a more complicated and costly manufacturing process in the present root fittings. The latter would include such processes as shot-peening and bearingizing of holes.

During the production of the main rotor blades, several changes were incorporated in the blade manufacturing to improve structural reliability and reduce manufacturing costs. Two of the changes were a change in adhesive material from AF6 to FM123 for a more reliable structural bond, and a change in spars from an extruded spar to a machined spar for better control of blade twist and contour.

Load/Life Comparison

The most significant loading parameters that establish the service life of the main rotor blade are gross weight, speed, rotor rpm and severity of manuevers. The design parameters for the prototype OH-6 were gross weight of 2100 pounds and a $V_{NE}$ of 128 knots. For the production OH-6A, the design parameters were increased to the following: gross weight = 2163 pounds at $V_{NE}$ = 130 knots; gross weight = 2400 pounds at $V_{NE}$ = 123 knots; and overload gross weight = 2700 pounds at $V_{NE}$ = 112 knots.

The main rotor rpm limits were the same for all configurations with the exception of the minimum rpm at 2700 pounds gross weight which was increased for power off (autorotation) from 400 to 465 rpm. The maximum limit load factor was reduced inversely with increase in gross weight, as the ability to obtain higher load factors is limited by the lift that can be developed by the main rotor blades.

The service life of the main rotor blade has remained constant at 1655 hours from the prototype OH-6 blades to the present in-service production OH-6A blades. The improvements in blade fatigue strength, mass balance and maintaining of airfoil contour, as previously stated, have

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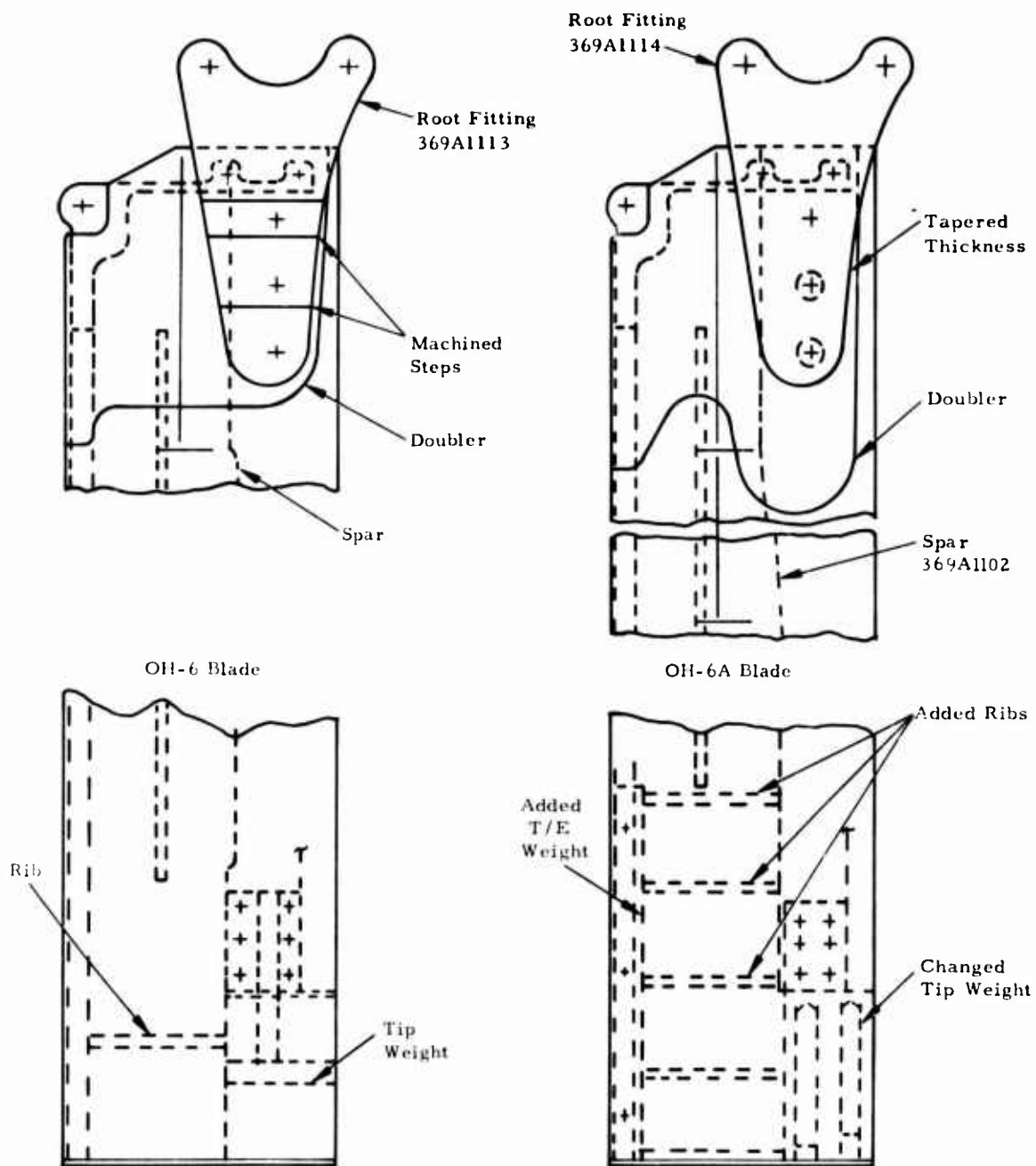
Figure 16. Main Rotor Blade Configuration.

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compensated for the increased performance of the OH-6A helicopter in gross weight, speed and maneuverability, resulting in no change in the main rotor blade service life.

Army field service reports have never reported a main rotor blade fatigue crack or failure. In excess of 1.5 million flight hours have been flown during this reporting period. The service experience substantiates the original design fatigue spectrum to be realistic.

## TAIL ROTOR BLADE

### History

The selection of the tail rotor blade configuration for the prototype OH-6 was based on the experience, knowledge and testing of the Model 269A tail rotor blade. The basic blade cross section was similar, consisting of a steel spar with the airfoil shape constructed of fiberglass and bonded to the steel spar. The chord was increased from 3.50 inches to 4.81 inches. The main structural modification for the prototype OH-6 tail rotor blade was in the root area in the method of attaching the blade to the tail rotor hub and retention strap pack. The spar was a straight tube partially formed to the airfoil section shape outboard of the root area. To obtain the final airfoil shape, it was necessary to add a resin filler material between the spar leading edge and fiberglass skin.

During the flight testing of the prototype OH-6, it was found that the filler material did not have sufficient strength to withstand the centrifugal force environment of the rotating tail rotor blade.

The major modifications for the production OH-6A tail rotor blades were to more closely shape the steel spar to the final airfoil shape, to bend the spar forward outboard of the root area and to change the tip weight configuration to allow balancing weights to be added or removed. The modifications resulted in eliminating the resin filler problem of the prototype OH-6 tail rotor blade and improving the blade mass balance. This resulted in a reduction of the cyclic loads imposed on the tail rotor blade.

The tip weight required for the production OH-6A tail rotor blade was initially estimated to weigh 86 grams, but after preliminary flight testing and additional analysis, it was determined that the tip weight could be reduced to 50 grams. The blade spar and retention system had been designed for the higher weight, and the reduction reduced the steady stresses in the blade spar. This effectively increased the blade spar fatigue strength.

During the production of the tail rotor blades, the only significant structural change was the addition of rivets attaching the tip cap to the blade fiberglass skin. The rivets were added in addition to the bonding adhesive providing a fail-safe attachment.

Load/Life Comparison

The major loading parameters, such as gross weight, speed, rotor rpm, load factors and engine power, are not reflected in the establishing of the service life for the tail rotor blade. With the exception of severe pedal reversal maneuvers, the only condition that produces fatigue damage is the ground-air-ground cycle.

The endurance limit resultant cyclic bending moment is 950 inch-pounds at the 7.2-inch radius of the tail rotor blade. A review of the fatigue load spectra presented in Table VI of Task II shows only one flight condition that would be fatigue damaging. Therefore, no information of value can be obtained from comparing tail rotor fatigue load spectra for flight conditions.

The service life of the prototype OH-6 tail rotor blade was 2598 hours. The service life for the production OH-6A blade increased to 2861 hours. The increase in service life was due primarily to the reduced tip weight which resulted in a lower ground-air-ground fatigue cycle.

DRIVE SYSTEM COMPONENTS

The OH-6A main and tail rotor transmissions were designed for minimum size and weight consistent with the U.S. Army requirement for a lightweight high-performance vehicle. The original design specification called for a 2160-to-2400-pound vehicle with an engine delivering 250 horsepower (HP) takeoff rating and 212 HP continuous. The original design specification power (torque) spectra for both transmissions, used during the engineering design and development phase of testing, were made obsolete early in pre-RVN operation when the OH-6A mission requirements were upgraded. Further upgrading took place when the actual RVN mission requirements were stated. The following paragraphs describe the historical changes that have taken place in the main and tail rotor transmissions from the time of original engineering development to the present. These changes have been analyzed for their cause-effect relationship with the upgraded OH-6A mission profiles as influenced by increases in gross weight and engine power.

## Design Criteria (and Certification)

### Main Transmission

The prototype and original production design criteria input power spectra for the main transmission (Reference 13 and 14) are virtually identical -- 248 horsepower takeoff for the prototype and 250 horsepower takeoff for original production. This spectrum, shown in Figure 17, develops a cubic mean power of 173 horsepower. The output torque spectrum is the same for both prototype and original production. Static torque requirements (References 13 and 14) increased 20 percent from prototype to production design. The initial design criteria fatigue loads are identical for both the prototype and original production designs. Both the prototype and original production design helicopters were FAA type certificated for 250 horsepower at takeoff and 212 horsepower maximum continuous engine shaft power. Current FAA type certification is for 278 horsepower at takeoff and 243 horsepower maximum continuous engine shaft power.

### Tail Rotor Transmission

The original production design criteria output torque spectrum for the tail rotor transmission called for higher power requirements than that for the prototype design (References 13 and 14). A comparison of the torque spectra for the original production design versus prototype design is shown in Figure 18. Limit static torque was also increased from 1385 pound-inches (67 HP)* for the prototype transmission to 2170 pound-inches (104 HP)* for the original production design transmission. The changes in the spectrum amount to a 24-percent increase in the cubic mean load while the static load was increased 56 percent. The design criteria fatigue loads for the original production design transmission shafting were considerably greater, and much more specific, than those required on the prototype design. In both cases, however, the fatigue loads were verified by flight test measurements.

## Configuration Changes (Design Improvements)

### Main Transmission

Constant upgrading of the main transmission load spectrum has resulted in numerous changes in design. The most significant design change took place on the output gearshaft assembly which required strengthening at a

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*At 100 percent $N_2$.
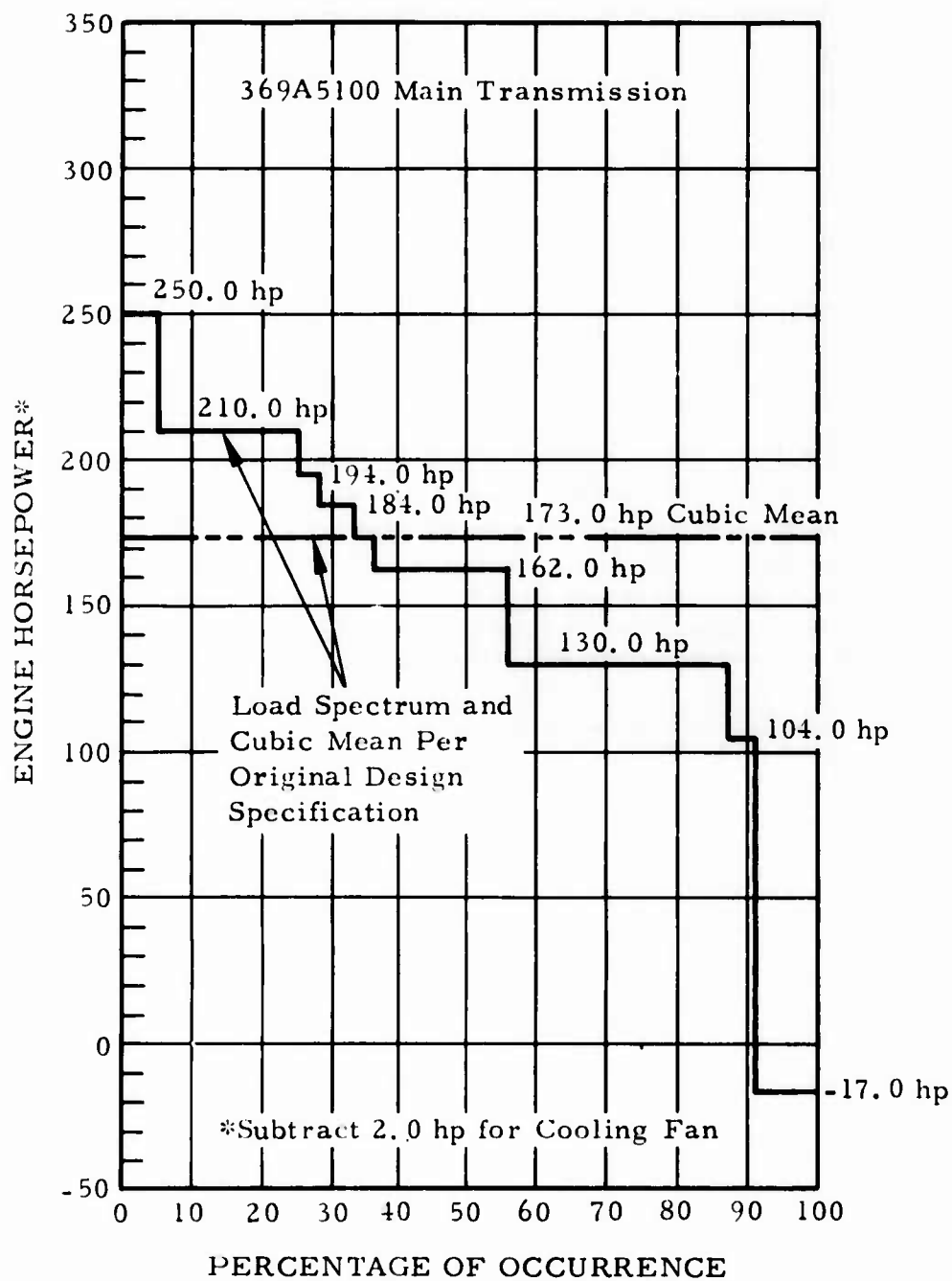
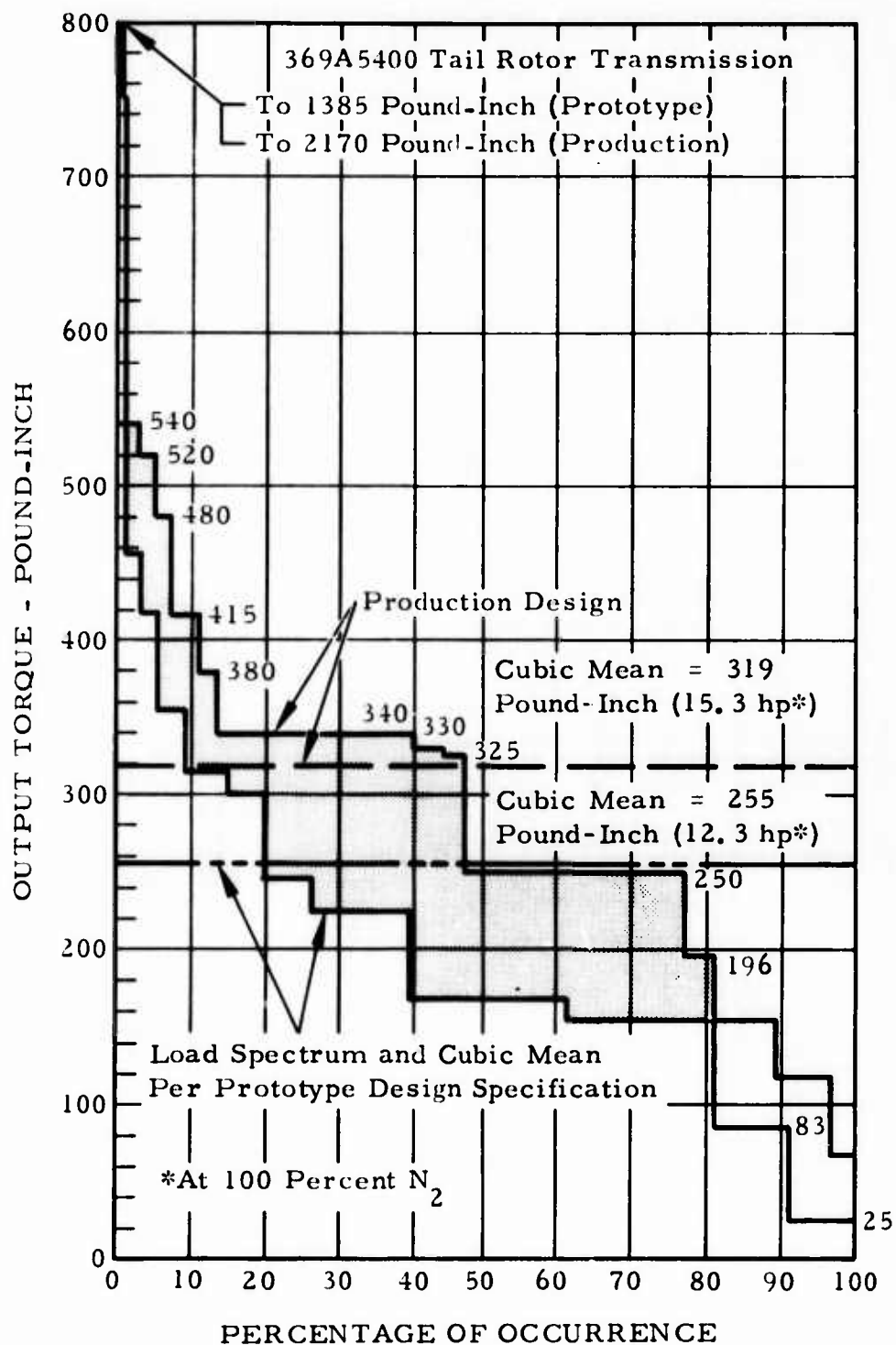Figure 17. Original Design Specification Input Power Spectrum.

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Figure 18. Comparison of Original Production Design Output Torque Spectrum Versus Prototype Design Specification Output Torque Spectrum.

fatigue critical section on the shaft. The details of the failure investigation and the design change are discussed in a later paragraph -- "Major Service Problems, Component Redesign." Other problems have included:

a. Bearing failures at less than $L_{B-10}$ life prediction

b. Spinning bearing races

c. Gear tooth wear

d. Electron beam weld failures

e. Inadequate lubrication.

Each of these service problems (and others) have been closely examined and corrective action taken to improve transmission reliability. Table XII shows the main transmission design changes that have taken place since the original production design transmission (369A5100-Basic) was released.

Tail Rotor Transmission

Operational service problems have brought about several design changes to the production tail rotor transmission. The most significant design change was concerned with strengthening the input gearshaft assembly, several of which failed in service. The details of the investigation and the design change are discussed later in this task. Bearing failures at less than $L_{B-10}$ life prediction make up the major portion of the continuing service problems. Table XIII shows the tail rotor transmission design changes that have taken place since the original production design transmission (369A5400-Basic) was released.

Load Spectra

Main Transmission

A 1200-hour tiedown endurance test, performed under a Product Improvement Program (PIP), was conducted on the production drive system between May 1966 and January 1967 (Reference 17). The cubic mean input power to the main transmission (369A5100-Basic) during the testing was 213 horsepower compared to 173 horsepower for the original production design requirement. A comparison of 1200-hour endurance test spectrum to the original design specification spectrum is shown in Figure 19.

| | | Helicopter Serial Number | |
| Configuration | Date | Effectivity | Remarks |
|---|---|---|---|
| Basic | September 1966 | 0001 thru 0711 | See drawing for serial number exceptions |
| -601 | October 1967 | 0001 thru 0711 | See drawing for serial number data |
| | | | Eliminate shim on pump drive to allow pump interchangeability |
| -603 | August 1968 | 0712 thru 0956 | Gear patterns specified |
| | | | Parco Lubrite Gears |
| | | | Acceptance Test Procedure specified |
| -605 | November 1968 | 0957 thru 1445 | Improved Lubrication and Higher Capacity Pump |
| | | | Hole in Scavenge Pump to lubricate output pinion roller bearing |
| | | | Higher capacity Taper Roller Bearing (tail rotor output) |
| | | | Improved bevel gear patterns |
| | | | Improved control of E. B. weld on output shaft |
| | | | Revised Parco Lubrite process |
| | | | Increased fit on roller bearing races |
| -607 | May 1970 | * | 369A5197 Input Pinion Roller replaced by 369A5180 Roller (M-50 steel) |
| | | | 369A5198 Output Pinion Roller replaced by 369A5198-3 Roller (M-50 steel) |
| | | | 369A5199 Output Shaft Roller replaced by 369A5199-3 Roller (M-50 steel) |
| | | | Loctite all roller bearing inner and outer races |

TABLE XII. 369A5100 MAIN TRANSMISSION DESIGN CHANGES

*All earlier configuration transmissions are up-graded to the -607 configuration at overhaul.

TABLE XIII. 369A5400 TAIL ROTOR TRANSMISSION DESIGN CHANGES

| Configuration | Date | Helicopter Serial Number Effectivity | Remarks |
|---|---|---|---|
| Basic | September 1966 | 0001 thru 0956 | |
| -601 | November 1968 | 0957 thru 1445 | 369A5431 Breather Assembly replaced by 369A5429 Breather Assembly |
| -607 | August 1972 | See Note | 369A5425-3 Input Gearshaft Assembly |
| -603 | January 1970 | Retrofit only when bearings are available | 369A5429 Breather Assembly replaced by 369A5429-3 Breather Assembly |
| | | | 369A5420 Output Pinion Roller replaced by 369A5432 Roller (M-50 material) |
| | | | 369A5423 Output Pinion Duplex replaced by 369A5433 Duplex (M-50 material) |
| -605 | August 1972 | See Note | Combines changes from (-607) and (-603) Configurations |

NOTE: All earlier transmissions are up-graded to the latest configuration at time of overhaul.

57
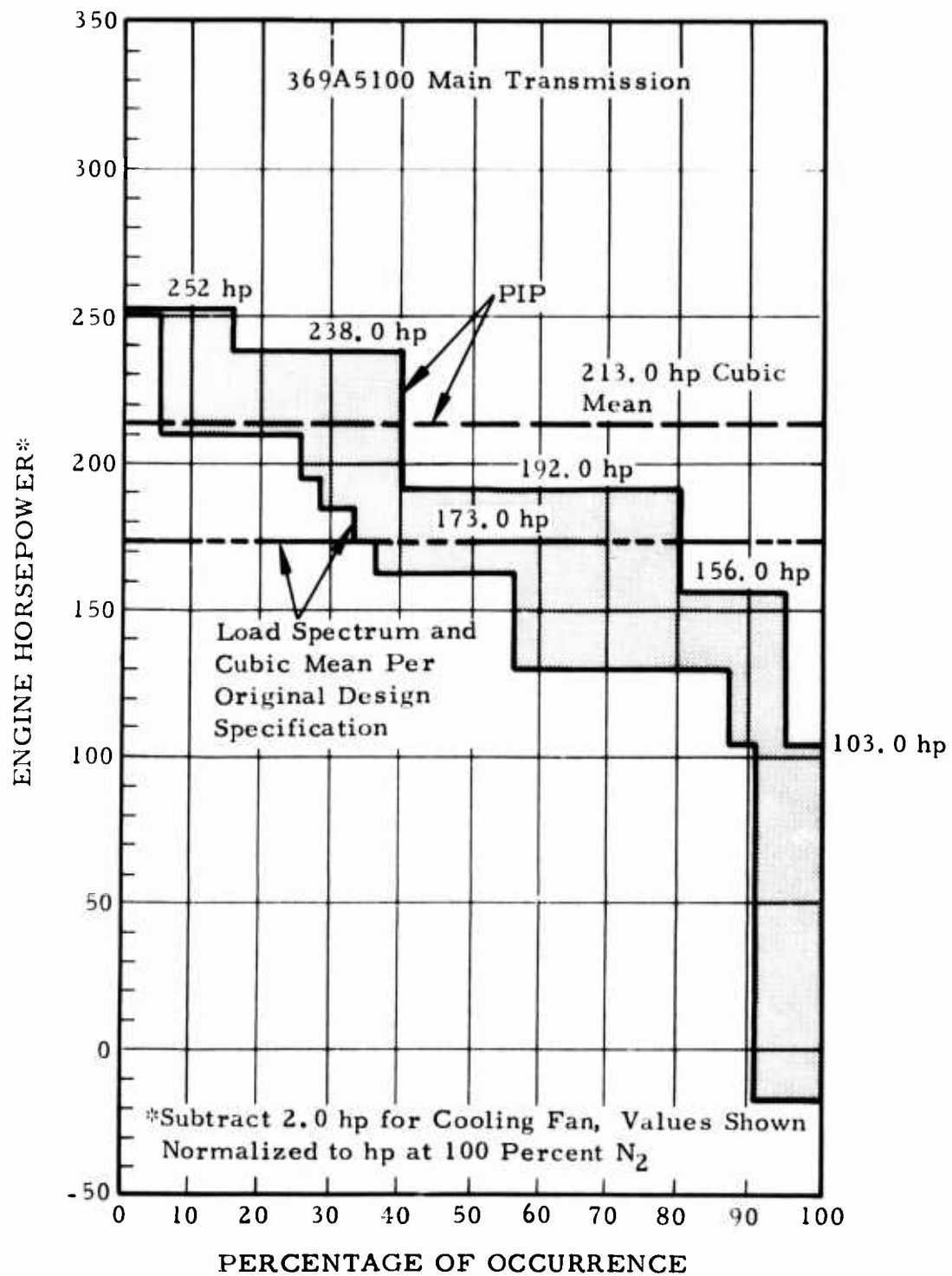
Figure 19. Comparison of PIP Input Power Spectrum Versus Original Design Specification Input Power Spectrum.

Additional endurance tests totaling 478 hours were conducted on the modified and upgraded 369ASK700 main transmission to the PIP spectrum shown in Figure 15 (Reference 21). This transmission, the prototype from which the 369A5100-605 transmission was derived, was also tested in a 10-hour ground run (Reference 23) to the loading spectrum shown in Figure 20.

A flight endurance program was also conducted on the 369ASK700 main transmission by the U.S. Army Aviation Test Board at Fort Rucker, Alabama, during 1968 (References 21 and 22). These tests, simulating RVN load conditions, accumulated 552 hours at 228 horsepower cubic mean with takeoff power at 270 horsepower flying at 2700 pounds gross weight. A comparison of the loading spectrum for these tests to the original design specification spectrum is shown in Figure 21.

A comparison of the measured load spectrum derived from the operational data (Reference 1) to the original design specification spectrum is shown in Figure 22. This comparison shows that the RVN power loadings were significantly higher that those anticipated by the original design specification. Of particular significance are the increased take-off and cubic mean power loadings.

Current testing is in progress to establish a takeoff rating of 317 horsepower, maximum continuous power at 235 horsepower and 233 horsepower cubic mean. This would then be compatible with the sea level standard rating of the Allison 250-C18 engine (Reference 18). A comparison of the loading spectrum for this testing versus the original design specification spectrum is shown in Figure 23.

Tail Rotor Transmission

Early in the program, a 100-hour endurance test (Reference 20) was conducted on the prototype design T/R transmission (369-5400). The cubic mean power for this test was 17.4 horsepower, compared to 12.3 horsepower for the prototype design specification and 15.3 horsepower for the original production design specification. A comparison of this endurance test spectrum to the prototype design specification spectrum is shown in Figure 24.

The original production T/R transmission (369A5400-Basic) was tested on the aforementioned 1200-hour endurance test. The cubic mean power to the tail rotor transmission during this test was 21.8 horsepower, compared
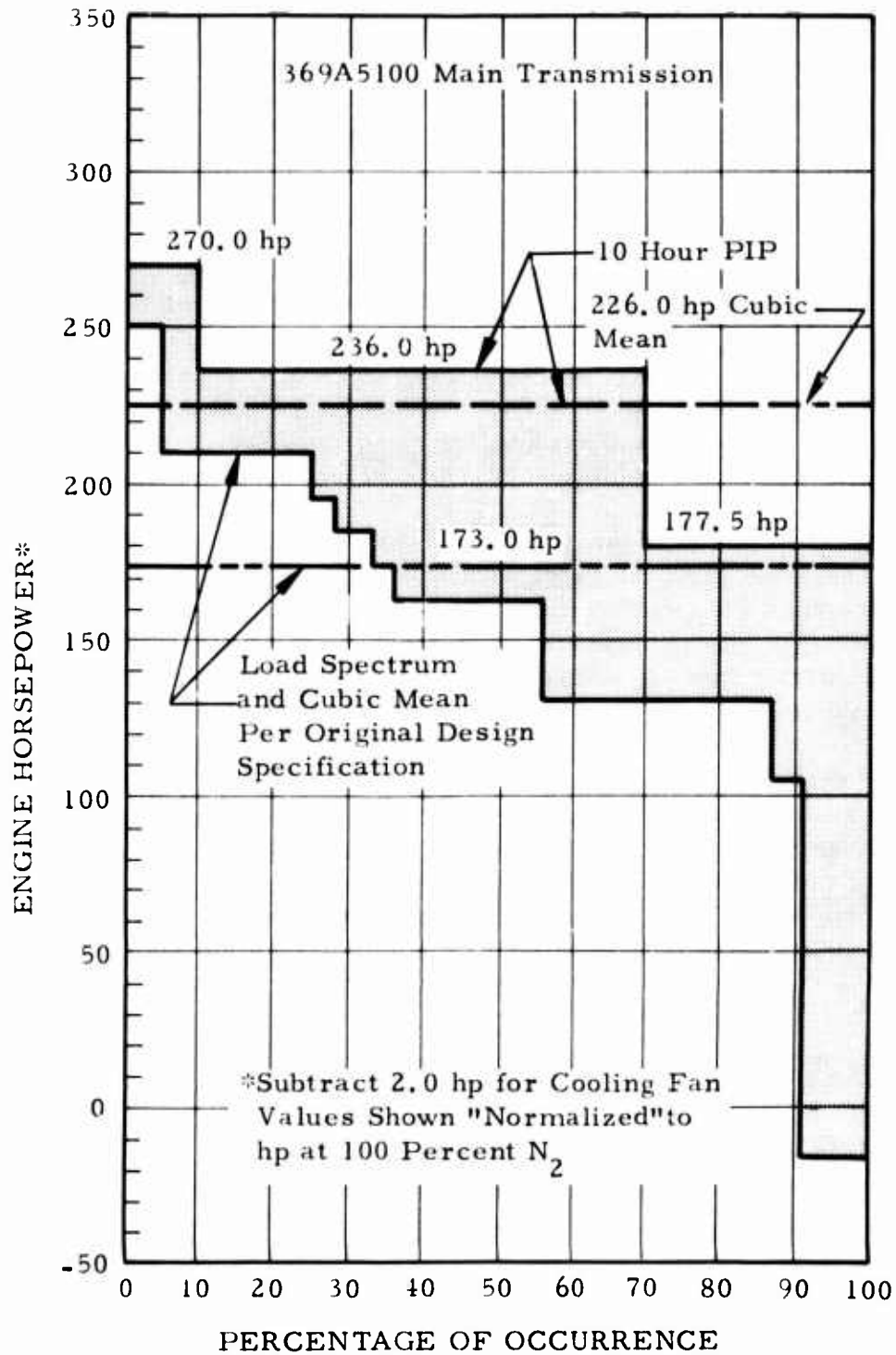
Figure 20. Comparison of 10 Hour PIP Test Input Power Spectrum
for the 369A5100-605 Main Transmission Versus Original
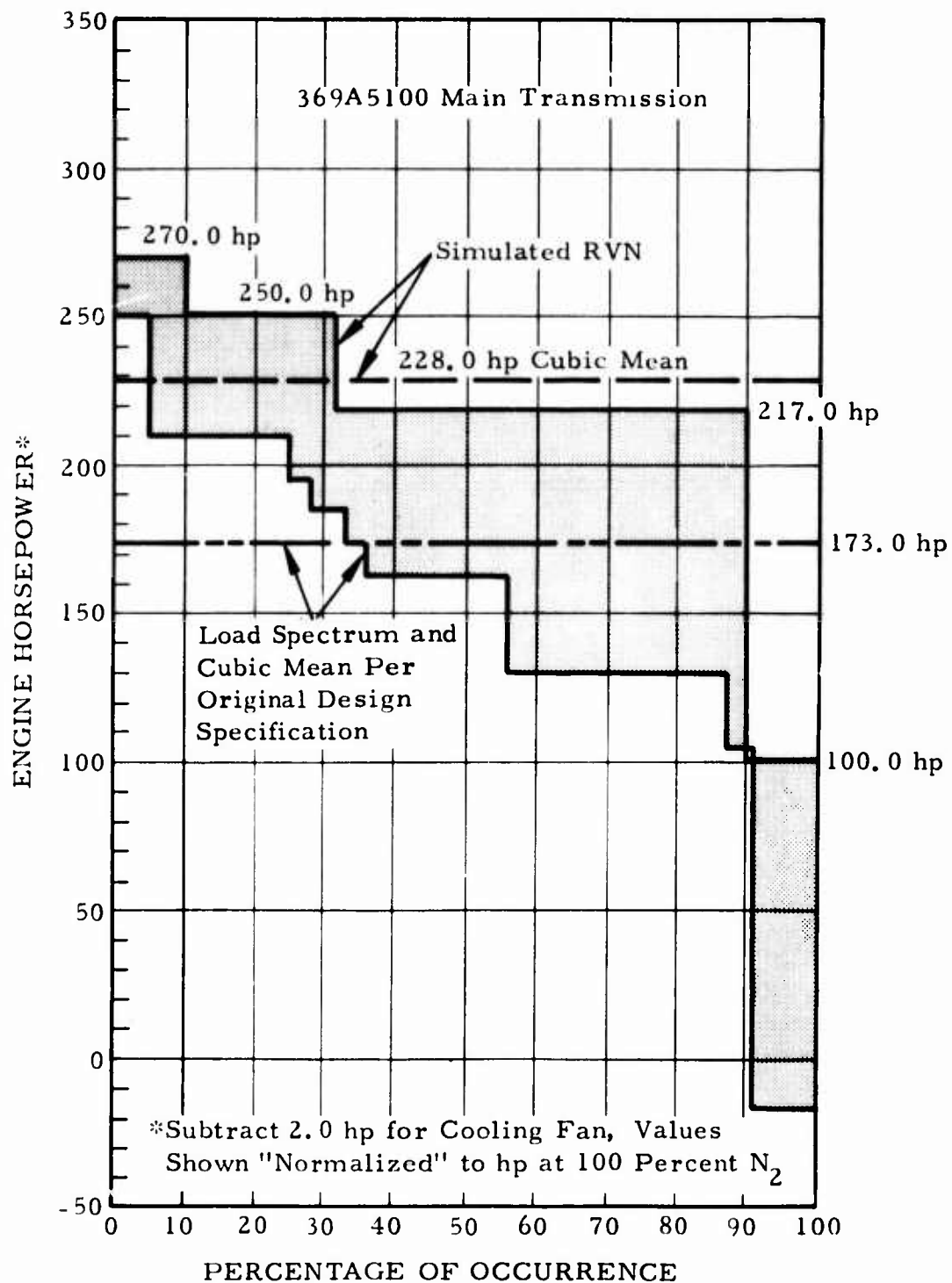Design Specification Input Power Spectrum.

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Figure 21. Comparison of a Simulated RVN Input Power Spectrum Versus Original Design Specification Input Power Spectrum.
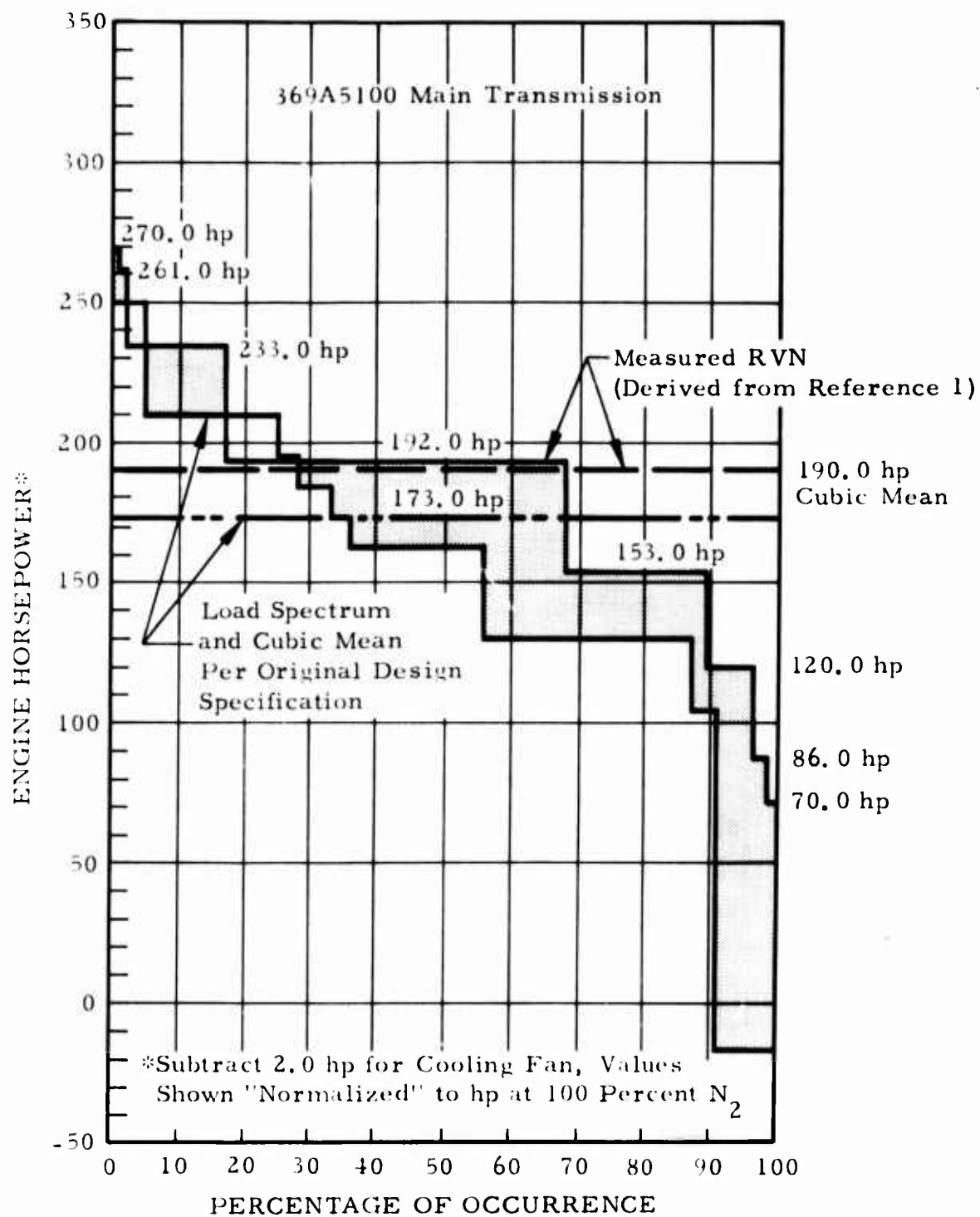
61

Figure 22. Comparison of Measured RVN Input Power Spectrum Versus Original Design Specification Input Power Spectrum.
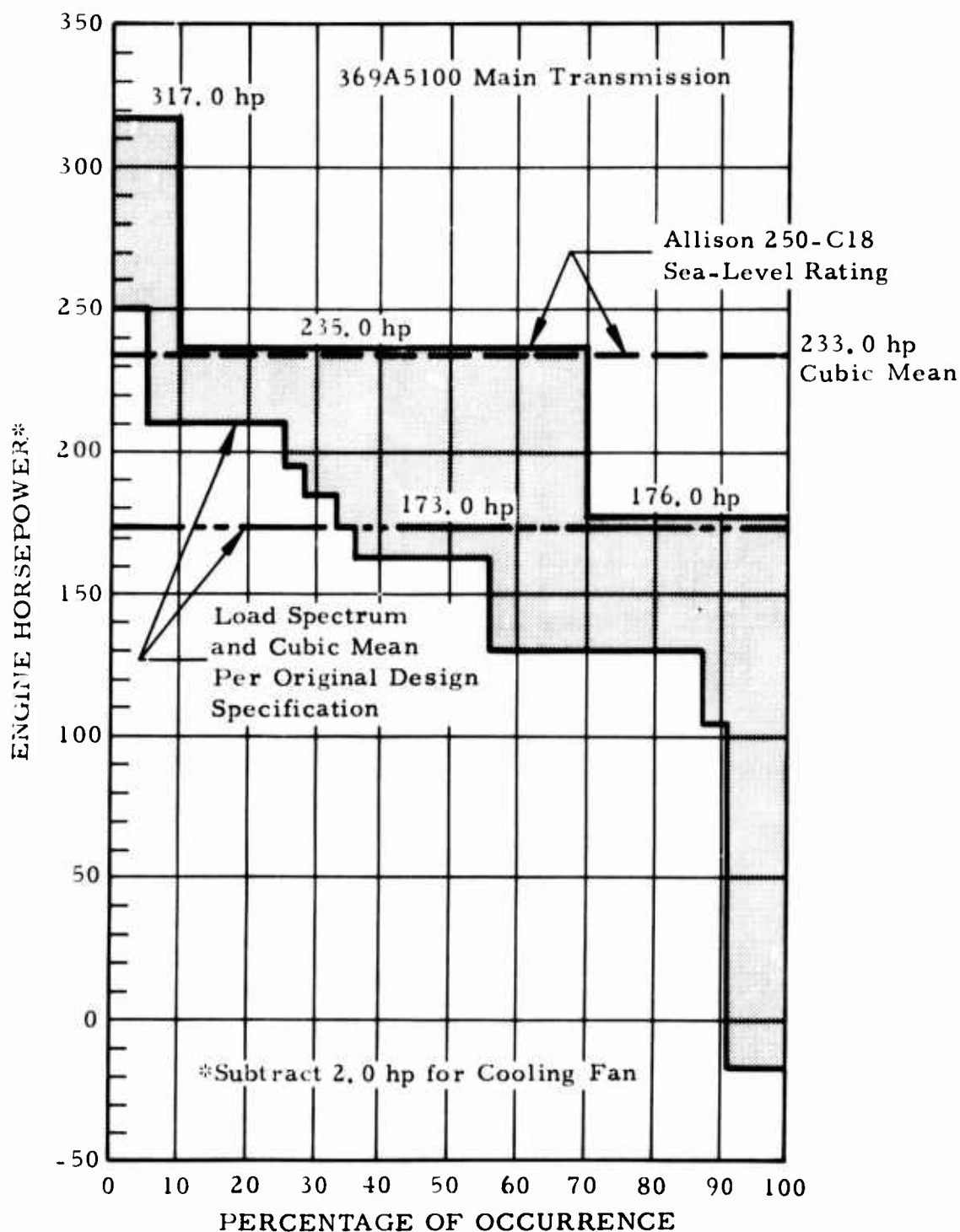
Figure 23. Comparison of Input Power Spectrum for the Allison 250-C18 Engine Sea Level Rating Installation Versus Original Design Specification Input Power Spectrum.

63
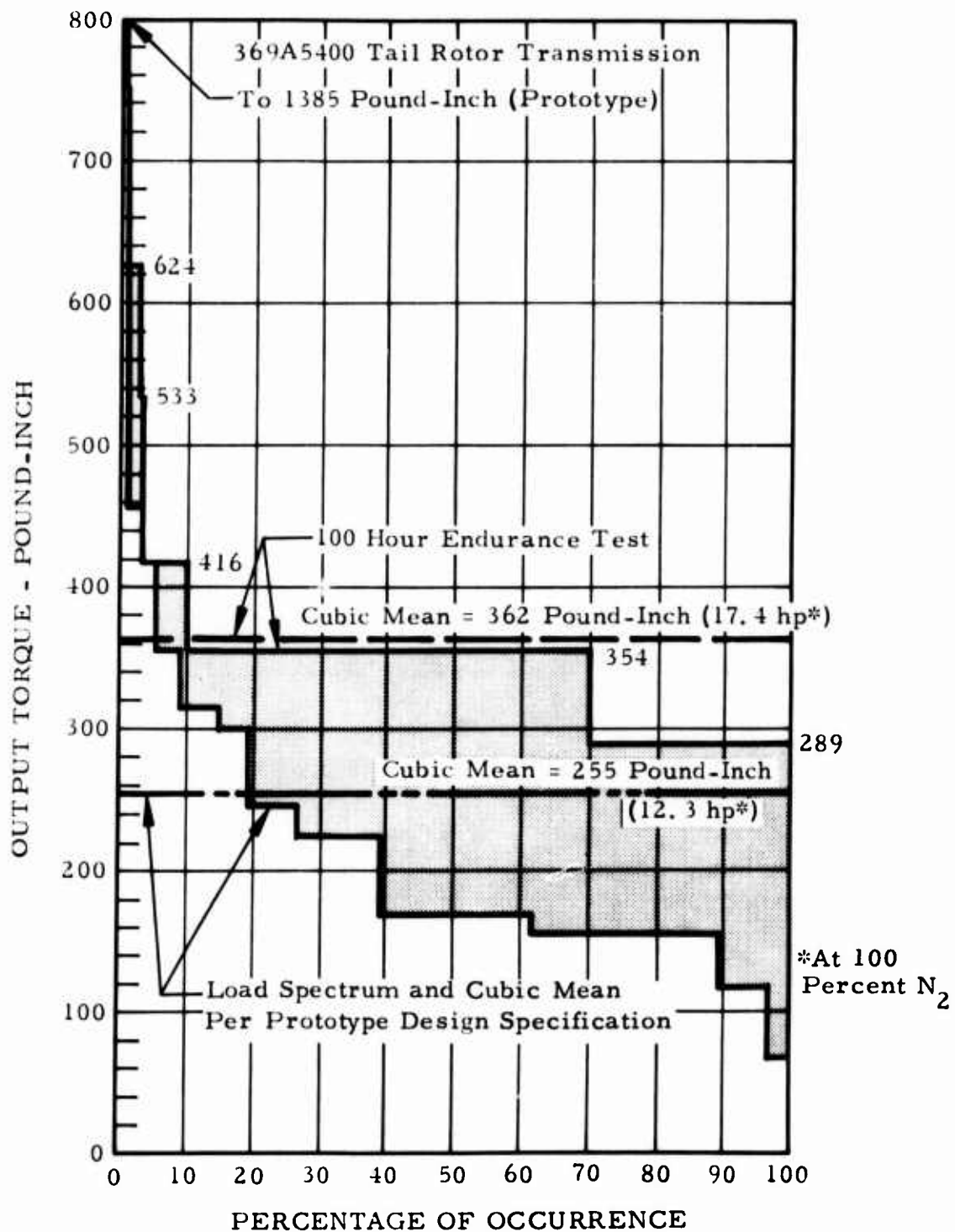
Figure 24. Comparison of 100-Hour Endurance Test Output Torque Spectrum Versus Prototype Design Specification Output Torque Spectrum.

to 15.3 horsepower for the production design specification. A comparison of this endurance test spectrum to the original production design specification spectrum is shown in Figure 25.

No operational loads data are available for the tail rotor transmission.

Major Service Problems - Component Redesign

Main Transmission

The output gearshaft assembly was originally considered to be an unlimited life part (Reference 15 and 16) under the loading conditions outlined in the original production design specification (Reference 14). However, during June and July 1968, three of the shafts failed in service, and examination of these parts indicated that a particular section of the shaft was more critical in fatigue than previously realized. While the specific cause of the failures was attributed to tears in the material due to poor machining practice, the high probability of occasional overtorque under combat conditions was also considered so that the subsequent design change also increased the wall thickness as well as improving the surface finish. Bench fatigue tests were performed on the (original) thinner wall shaft design and a limited life (39,800 hours) was assigned to these parts based on measured flight loads data compared to the design S-N curve derived from the bench tests (Reference 16, Appendix H). Fatigue analysis and the tests indicate, however, that the improved, thicker wall shaft design is an unlimited life component.

The failure investigation also revealed that, although the critical section of the shaft had been strengthened between prototype and production design, the improvement was necessary to completely eliminate the cause of the failures. The design changes, from prototype to original production to improved production, are shown in Figure 26.

Tail Rotor Transmission

Occasional failures of the input bevel gearshaft assembly in service were initially attributed to rough surface finish and to tool marks found at the most critical section in the failed shafts (Figure 27). However, when similar failures subsequently occurred in smoothly machined specimens, further investigations were conducted to determine the cause.
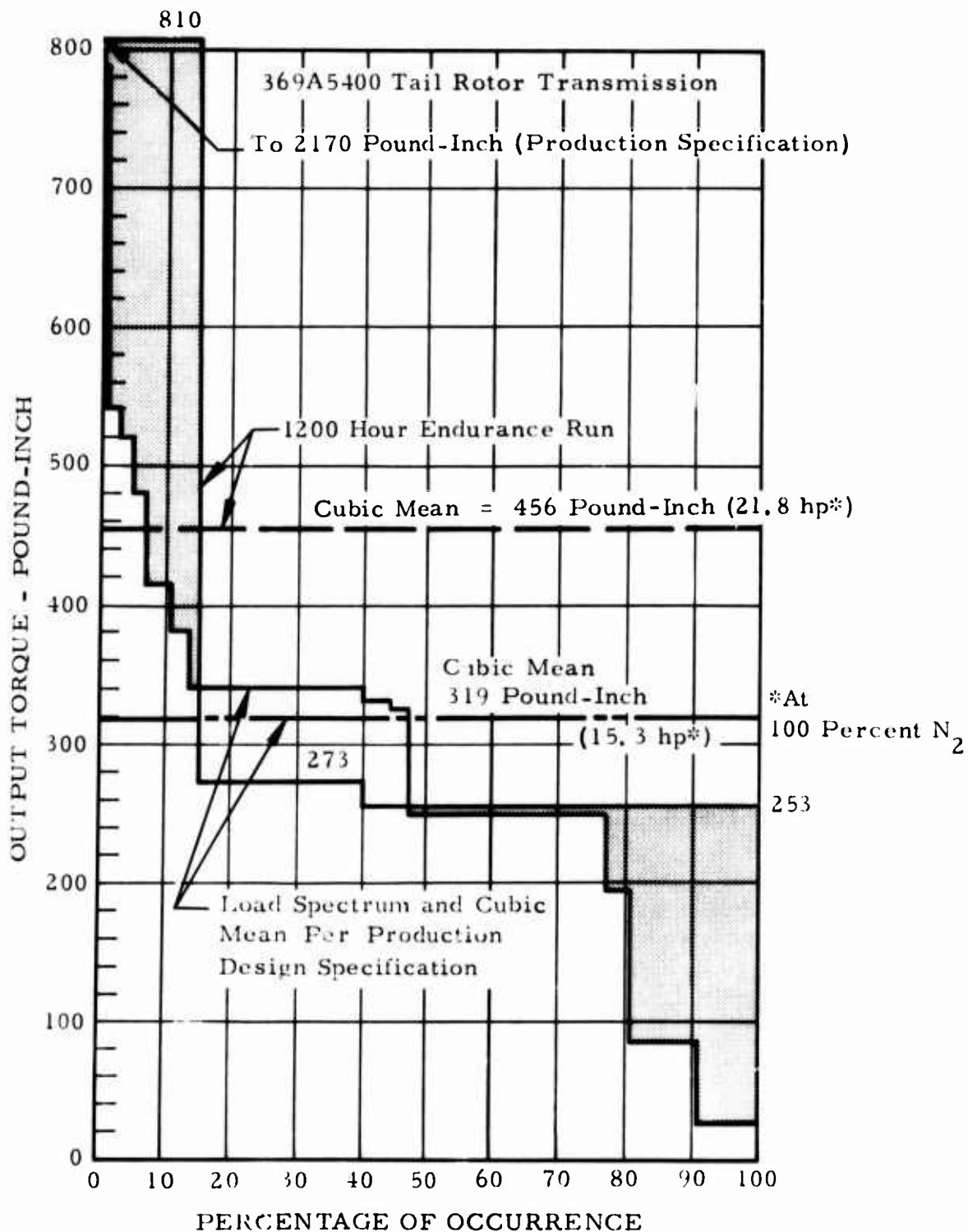
65

Figure 25. Comparison of 1200-Hour Endurance Run Output Torque
Spectrum Versus Original Production Design Specification
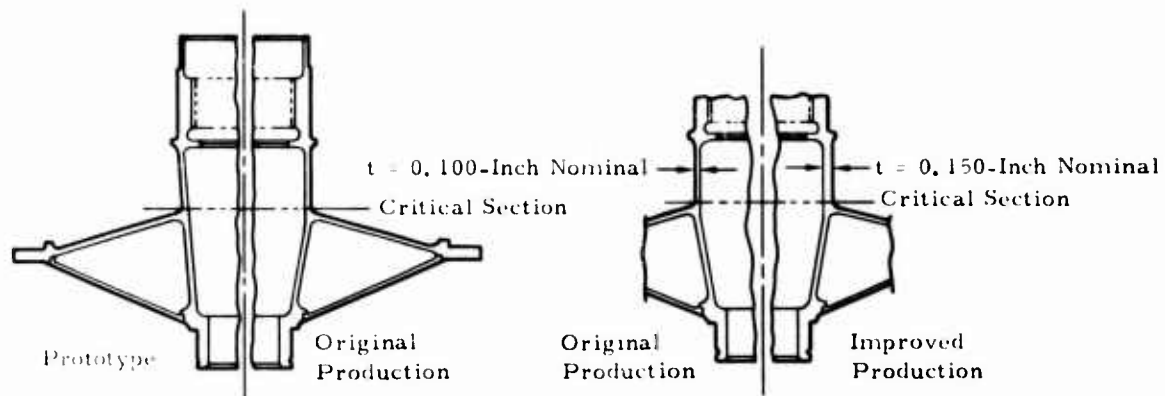Output Torque Spectrum.

Figure 26. Design Improvements to the Critical Fatigue Section on the 369A5158 Main Transmission Output Shaft Assembly.
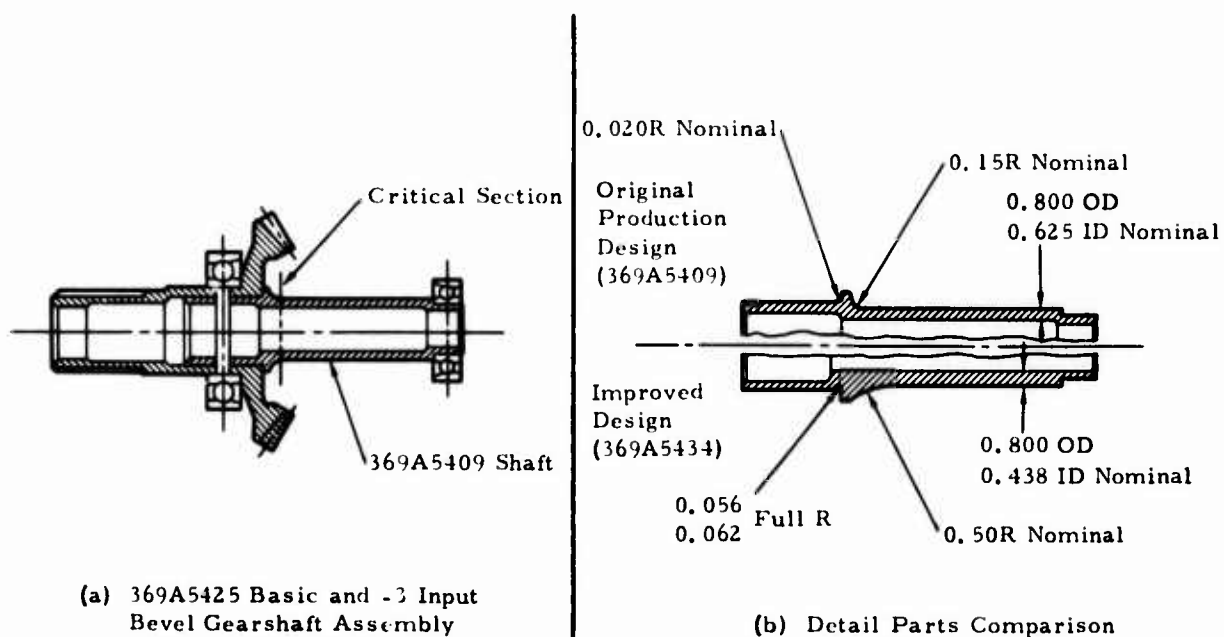


(a) 369A5425 Basic and -3 Input Bevel Gearshaft Assembly

(b) Detail Parts Comparison

Figure 27. Design Improvements to the Critical Fatigue Section on the 369A5425 Input Bevel Gearshaft Assembly.

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An in-depth investigation of the failures, reported in Reference 19, revealed that the shaft assembly is more highly loaded than previously anticipated, particularly under certain maneuvers. The flight strain data in Reference 19 indicates that certain pedal reversal and sideward flight maneuvers conducted in the original OH-6A flight test spectrum were not as severe as encountered in combat operation. The critical conditions are hovering pedal reversals and right sideward flight. The input shaft was redesigned to compensate for the higher operational loading.

The geometry of the improved design, as compared to the original design, is shown in Figure 27. In addition to increased wall thickness, larger fillet radii and improved surface finish, the shaft material was upgraded from air melt 4340 steel, heat treated to $R_c$ 30-35, to consumable electrode vacuum melt 4340, heat treated to $R_c$ 36-40. These improvements substantially increase the fatigue life of the shaft assembly.

The improved design part is interchangeable with the original design part and is suitable for retrofit at component overhaul. New gearboxes are now having the change incorporated.

## Hughes Flight Tests

The original flight test program (on the prototype YOH-6A) was conducted in 1963. The loads data obtained in these tests (Reference 7) were used in establishing the original service lives of the OH-6A rotor drive system components. Additional flight test programs were conducted in 1967 (Weapons System Installation - Reference 18) and 1969 (Increased Gross Weight Certification - Reference 24). Current (1972) service lives of the OH-6A rotor drive system components are based on the cumulative data obtained in these test programs. A summary of the input power/gross weight configurations used in the test programs is shown in Table XIV.

## Critical Components Service Lives

Tables XV and XVI present service lives for historical changes of the main and tail rotor transmission limited life components.

## Cause-Effect Analysis

Overall review of the historical data reveals considerable indication that both the main rotor and tail rotor drive trains have been subjected to higher loads (torque) than anticipated by the original design specification, especially during severe maneuver conditions.

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TABLE XIV. HUGHES FLIGHT TEST PROGRAMS INPUT POWER/GROSS WEIGHT DATA

| Item | Reference 7 (1963) | Reference 8 (1967) | Reference 24 (1969) | |
|---|---|---|---|---|
| Engine Torque Pressure | Takeoff | 75 psi | 75 psi | 80.3 psi | 80.3 psi |
| | Maximum Continuous | 63.5 psi | 63.5 psi | 70 psi | 70 psi |
| Gross Weight | | 2100 lb | 2400 lb | 2600 lb | 2100 lb |

TABLE XV. HISTORICAL CHANGES - MAIN TRANSMISSION LIMITED LIFE COMPONENTS

| Component | Prototype | | OH-6A Production | | |
|---|---|---|---|---|---|
| | Part Number | Life (Hours) | Part Number | Life (Hours) | Transmission Serial Number Effectivity |
| Output Gearshaft Assembly | 369-5158 | Unlimited | 369A5158 | 39,800 | 0001 thru 1537 |
| | | | | Unlimited | 1538 and subsequent |

| Component | Part Number | Life (Hours) | Remarks |
|---|---|---|---|
| Output Pinion Gearshaft Assembly | 369-5408 | 3730 | Prototype only - part of 369-5406 gear set. |
| | 369A5408 | 7180 | Original production service life based on prototype flight strain survey. |
| | | 2610 | Superseded original production service life - includes extended envelope flight strain survey. |
| | | 2940 | Correction of previous service life calculations. |
| Input Bevel Gearshaft Assembly | 369-5407 369-5409 | 3730 | Prototype only - part of 369-5406 gear set. |
| | 369A5425 (Basic) (-3) | 7180 2610 2940 | See remarks above and note |
| | | 1800 | Based on a newly developed design S-N curve same flight strain surveys above. |
| | 369A5425 (-5) | 27,600 | Based on improved shaft, design S-N curve and same flight strain surveys above. |

**TABLE XVI.  HISTORICAL CHANGES - TAIL ROTOR TRANSMISSION LIMITED LIFE COMPONENTS**

NOTE:   The original service lives of the input bevel gearshaft assembly were based solely on matched set replacement and are the same as calculated for the output pinion gearshaft assembly.

## Mission Profile

Comparison of the original design specifications for the OH-6A (Reference 14) to the operational flight loads data (Reference 1) indicates that the RVN operations required more time at higher powers and higher gross weights than anticipated. According to Figure 4 of Reference 1, some 29 percent of flight time was spent at gross weights exceeding 2400 pounds, the maximum gross weight of the original design specification. Figure 5 of Reference 1 indicates that 53 percent of ascent time (6 percent of total time) was at gross weights exceeding 2400 pounds. Figure 10 of Reference 1 indicates that 2 percent of the time was spent at engine torques above 75 psi (takeoff) and 14 percent spent at engine torques above 63.5 psi (maximum continuous power). The significance of these factors is reflected in Figure 22 of this report, which shows a comparison of the RVN power data, developed from Figures 4, 5, 6 and 10 of Reference 1, to the input power spectrum for the original design specification for the main transmission. The engine torque pressure data presented in Figure 10 of Reference 1 is converted to engine horsepower by the following equation:

$$\text{Engine Horsepower} = \left(\frac{\text{Torque Pressure}}{0.3}\right)\left(\frac{\% \, N_2}{100\%}\right) + 2.5$$

where:

   0.3 = Torque Pressure Conversion Factor (per Allison
            250-C18 Engine Specification)

   $N_2$ = Engine Output Shaft Speed, rpm

   2.5 = Engine Installation Adjustment Factor, HP

Engine torque pressure, however, is not a reliable indicator of shaft torque in other parts of the drive train. This is particularly the case with regard to maximum transient maneuver conditions and to tail rotor maneuver power requirements. For this reason, the RVN torque pressure measurements published in Reference 1 cannot be compared to the developmental power requirements except on an overall spectrum basis as shown in Figure 22.

Major Service Problems

The service failures of the main transmission output gearshaft assembly and the tail rotor gearbox input gearshaft assembly cannot be attributed solely to manufacturing or metallurgical defects since flight tests (Reference 19) have confirmed that higher transient loads (torques) will occur during combat operational-type maneuvers of the helicopter. Also, some of the service failures occurred on the shafts with no detectable defects. This type of data is not available in Reference 1.

The flight test data loads of Reference 7, 8, 19 and 24 indicate that while increased gross weight and increased power both have influence on steady torque, the most damaging fatigue loads, under all gross weight and power conditions, are the transient peak loads that generally occur during maximum maneuver or power failure conditions. This is particularly true of the tail rotor transmission components which, under maximum g pullup, pedal reversal and sideward flight conditions, experience transient torques more than twice the normal steady-state loads. It is these transient peak loads rather than the steady-state conditions that limit the lives of the rotor drive system components.

# TASK IV - ANALYSIS OF PARAMETER PEAK VALUES

## INTRODUCTION

A comparison is presented between the peak values of the parameters recorded during the program reported in Reference 1, the OH-6A design criteria maximums and the highest values encountered during Hughes conducted flight test programs. The parameters included in the study were airspeed, vertical acceleration, longitudinal cyclic control position, collective control position, engine torque pressure and main rotor rpm.

## ANALYSIS

### Operational Peak Values

The peak values for the OH-6A operational data (Reference 1) are presented in Table XVII. The figures shown were obtained from the appropriate tables or plots in the referenced document. Unfortunately, the data are in blocks or ranges, and absolute one-time maximums or minimums cannot be determined. This factor has been detrimental in other tasks of this project, but not to the extent involved here. Knowledge of the actual peak values would be more helpful for comparison purposes than merely knowing the range. Further analysis of the oscillograph records would be required to determine the absolute maximums and minimums.

The particular mission segment during which the peak values were most predominant is also shown in Table XVII.

### Design Criteria Peak Values

The final column on Table XVII presents the structural design values for the parameters of interest.

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TABLE XVIII. PEAK VALUE COMPARISON

| Parameter | Operational Data | | Development Testing | | | | Design Criteria | |
|---|---|---|---|---|---|---|---|---|
| | Peak Value(s) | Mission Segment* | Peak Value(s) | Condition | Mission Segment | Peak Value(s) | Remarks | Basis |
| 1. Airspeed ~ knots | >124 (<130) | Descent | 141** | Autorotation and power on 111% VNE (= 111% VH) | Steady State | 130 | VNE (sea level) | Canopy Design |
| | | | | | | 144** | 111% VNE (sea level) | Rotor System Design |
| 2. Vertical Acceleration ~ g | >2.20 (<2.4) | Maneuver | 2.27 | Pullup, 2.25 g turn (2100 lb) | Maneuver | 2.91 | Pullup, 2.35 g turn (2100 lb) | Rotor System Design |
| | | | 1.69 | Pullup, 2.18 g turn (2400 lb) | Maneuver | 2.55 | Pullup, 2.06 g turn (2400 lb) | |
| | <0.20 (>0.0) | Descent | 0.27 | Entry to autorotation | Maneuver | -0.50 | | |
| 3. Longitudinal Cyclic Position ~ % from full aft | >90 | Steady State[1] | 83 | Forward flight | Steady State | 100 | Total stick travel | Center of gravity range |
| | (<10) | Steady State[1] | 5 | Rearward flight | Steady State | 0 | Total stick travel | Forward and rearward airspeeds |
| 4. Collective Position ~ % from full down | 80 to 90 | Steady State[1] | 87 | 111% VNE (= 111% VH) | Steady State | 100 | Total stick travel | Maximum speed |
| | 10 to 20 | Steady State[1] | 0 | Autorotation | Steady State | 0 | Total stick travel | Gross weight / Rate of descent |
| 5. Engine Torque Pressure ~ psi | >80 | Maneuver | 82 | 111% VNE (= 111% VH) | Steady State | 80.3 | Takeoff power | Engine and Drive System Design |
| 6. RPM | >490 | All | 545** | Autorotation, VNE | Steady State | 514 (540)** | Autorotation Redline | Hub Design |
| | <440 | Descent | 375** | Autorotation, VNE | Steady State | 400 (380)** | Autorotation Redline | |

* Of greatest occurrence
** Demonstration points
(1)Data shown for this mission segment only

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## Development Test Peak Values

The peak values encountered during Hughes development of the OH-6A Helicopter are shown in Table XVII. The parameters were recorded during flight strain surveys and Type Inspection tests that are required for FAA certification. The references involved are numbers 7, 8, 26, 27, and 28. The particular flight condition and mission segment that produced the peak value are noted.

## Comparison of Peak Values

Generally, the operational data and the developmental testing data show good agreement. The exceptions are airspeed and rpm where the operational peak values are the lesser, due to the fact that testing was conducted by Hughes during OH-6A development to verify the design criteria demonstration points* for those parameters. The peak values recorded during development testing were an airspeed of 141 knots, and 545 and 375 rpm for maximum and minimum red-line rpm, respectively. The corresponding operational values were greater than 124 knots (less than 130 knots) and greater than 490 rpm and less than 440 rpm.

Neither set of flight data produced load factors close to the design maximum g pullup (2.91) nor the minimum g value (-0.50). The turn load factors appear to compare favorably. The table shows a design criteria maximum of 2.35g and the peak Hughes flight strain value of 2.25g. The exact nature of the maneuver performed to obtain the operational peak of greater than 2.2g (less than 2.4g) cannot be ascertained from the data presented in the report. Additional analysis of the oscillograph records would be required to determine the flight conditions.

The longitudinal cyclic and collective control positions show some margin as compared to the design absolute control stops. This is to be expected because of the good control power characteristics of the OH-6A. Operational control position data are shown for the steady-state mission segment only because peak values are not presented for the other mission segments in Reference 1. The peak engine torque pressure values show satisfactory agreement for all three data sources.

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*(1) 111% x $V_{NE}$

(2) 105% x Maximum Power Off rpm

(3) 95% x Minimum Power Off rpm

## Analysis of Peak Values on a Probability Basis

In order to obtain a more meaningful comparison of selected parameter peak values, an analysis of the operational test data was performed on a probability basis. The maximum (or minimum) values recorded during the relatively short period (216 hours) involved in data acquisition are related to the probability of occurrence of greater values over a period more representative of a fleet's service life. The period selected was one million hours. The parameters studied include maximum and minimum load factors, maximum airspeed and maximum engine torque pressure.

The results of the study are presented in Figures 28 and 29. The data show that in all cases the parametric peak value for large values of fleet service are very likely to exceed the peak value measured during the 216 monitored flight hours by a significant margin.

Figure 28 shows the maximum and minimum load factor exceedance data. The points are taken directly from Reference 1, Figure 12.a. Extrapolating to a million hours, the figure shows the maximum load factor expected is 3.0g while the predicted minimum value is -0.8g. The peak values recorded during the operational data acquisition period were 2.2 to 2.4g and 0.2 to zero g for the maximum and minimum cases, respectively. Reference 1 presents load factor exceedance data for several gross weight ranges (Figure 28 of this report is a composite of all weights). The data of Reference 1 show that the peak load factors are more likely to occur at gross weights less than 2200 pounds.

Table XVIII shows the conversion of the airspeed and engine torque data presented in Reference 1 to a form suitable for this analysis. After the composite number of hours spent in each range was determined from all four mission segments, the number of occurrences within that time was calculated by assuming a duration of three seconds for each flight condition which would count as an occurrence. The number of flight hours required to attain a single occurrence in any range was then determined by dividing the number of occurrences into the total time spent in all ranges. Figure 29 is a graphical presentation of the data developed from the foregoing discussion. As can be seen from the figure, the maximum airspeed and engine torque expected within one million hours are 150 to 180 knots and 104 to 130 psi, respectively. The comparable operational values were 120 to 130 knots and greater than 80 psi. Due to the magnitude of the extrapolation involved, ranges of predicted values were selected. In each case the upper or solid curve on Figure 29 represents a linear fairing resulting in what is considered to be a conservative peak value at one million hours. Additional operational test time would provide much more reliable predicted peak values.
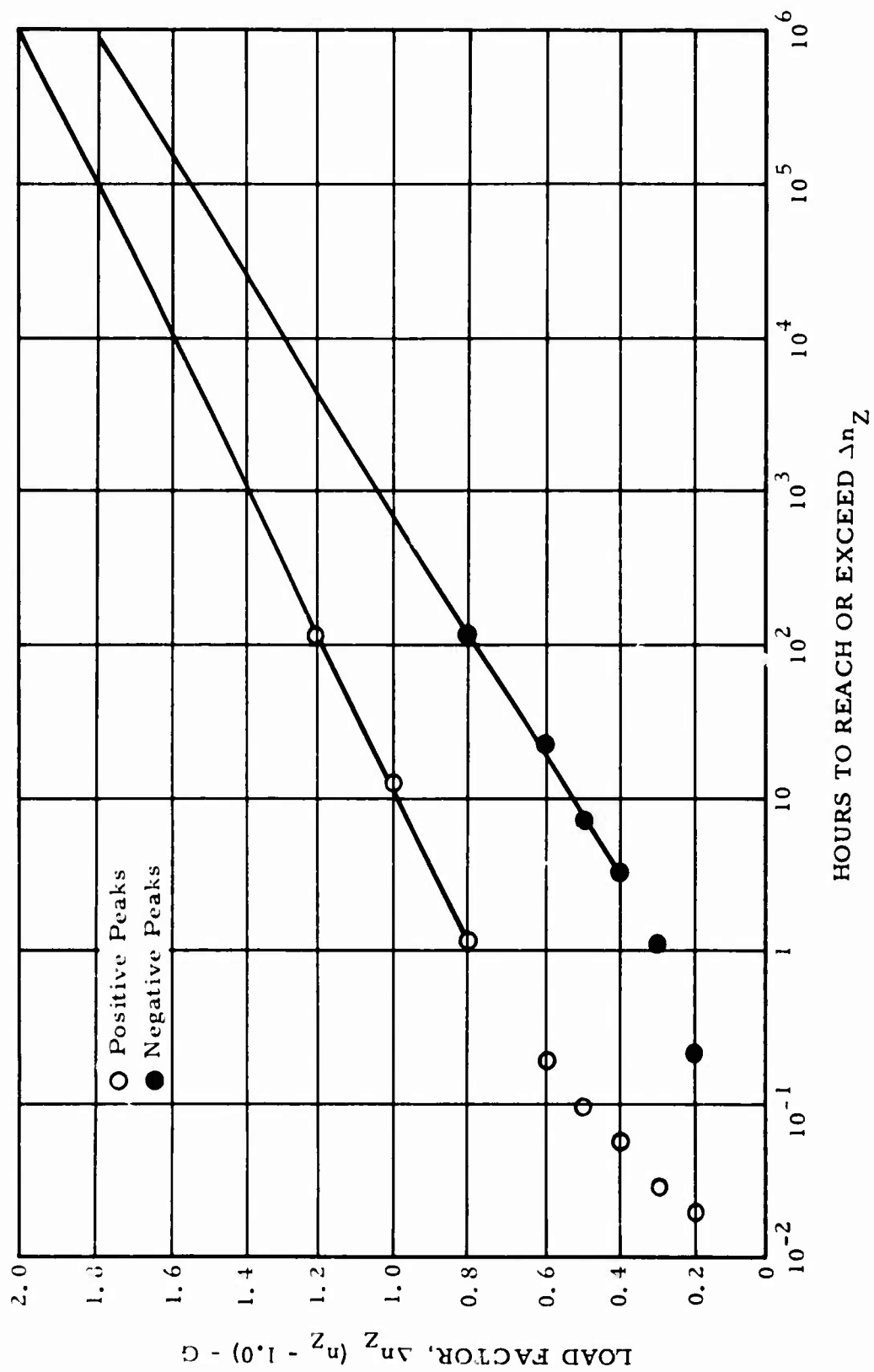
HOURS TO REACH OR EXCEED $\Delta n_Z$

LOAD FACTOR, $\Delta n_Z$ $(n_Z - 1.0) - G$

○ Positive Peaks
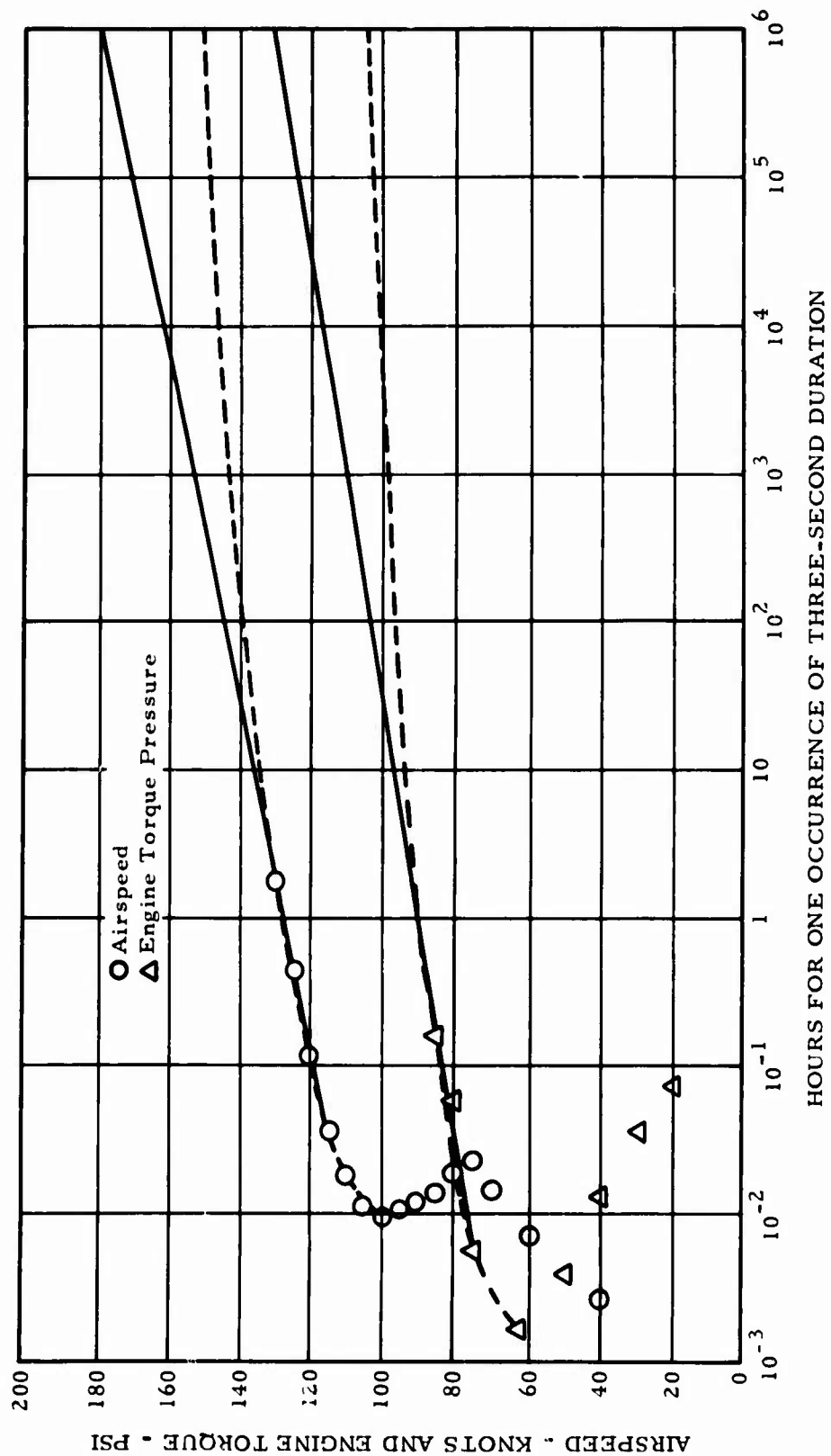● Negative Peaks

Figure 28.  Load Factor Exceedance.

Figure 29.   Airspeed and Engine Torque Exceedance.

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TABLE XVIII.   AIRSPEED AND ENGINE TORQUE EXCEEDANCE

| Airspeed Range ~ Knots | Hours* Spent in Range | Number of 3-Second Duration Occurrences | Flight Hours to Attain 3-Second Occurrence | ETP Range ~PSI | Hours* Spent in Range | Number of 3-Second Duration Occurrences | Flight Hours to Attain a 3-Second Occurrence |
|---|---|---|---|---|---|---|---|
| 0 to 40 | 64.5 | 77,400 | 0.0028 | 0 to 20 | 2.50 | 3,000 | 0.0723 |
| 40 to 60 | 24.6 | 29,520 | 0.0073 | 20 to 30 | 5.00 | 6,000 | 0.0362 |
| 60 to 70 | 13.0 | 15,600 | 0.0138 | 30 to 40 | 15.40 | 18,480 | 0.0117 |
| 70 to 75 | 7.8 | 9,360 | 0.0230 | 40 to 50 | 45.60 | 54,720 | 0.0040 |
| 75 to 80 | 9.4 | 11,280 | 0.0191 | 50 to 63.5 | 113.50 | 136,200 | 0.0016 |
| 80 to 85 | 13.4 | 16,080 | 0.0134 | 63.5 to 75 | 30.70 | 36,840 | 0.0059 |
| 85 to 90 | 14.2 | 17,040 | 0.0126 | 75 to 80 | 3.03 | 3,636 | 0.0600 |
| 90 to 95 | 16.6 | 19,920 | 0.0108 | 80 to 85 | 1.23 | 1,476 | 0.1470 |
| 95 to 100 | 18.8 | 22,560 | 0.0095 | | | | |
| 100 to 105 | 16.1 | 19,320 | 0.0111 | | | | |
| 105 to 110 | 9.6 | 11,520 | 0.0187 | | | | |
| 110 to 115 | 4.8 | 5,760 | 0.0373 | | | | |
| 115 to 120 | 1.6 | 1,920 | 0.1119 | | | | |
| 120 to 124 | 0.4 | 480 | 0.4477 | | | | |
| 124 to 130 | 0.1 | 120 | 1.7908 | | | | |

*Based on Reference 1, Figures 10 and 11.

## OH-6A Limiting Characteristics

An evaluation of pilot flight cards, test reports, oscillograph data and pilot interviews was conducted to define the OH-6A characteristics that influence the magnitude of the load parameters discussed in the previous section. Table XIX lists each of the parameters and the characteristics that limit the peak values.

The two control positions, longitudinal cyclic and collective, differ from the other parameters inasmuch as the design limiting characteristics exist as physical stops and not as flight manual limitations that can be exceeded. Whereas the pilot can exceed the redline limitations of airspeed, engine torque pressure and rpm by not observing the appropriate cockpit instruments, the same is not true for the control positions where contacting a stop represents the absolute limit of motion.

The most significant limiting characteristic is retreating blade tip stall which affects several of the parameters. Maximum airspeed, maximum load factor and minimum rpm are all primarily affected by retreating blade tip stall.

Design features of the OH-6A helicopter result in many of the characteristics listed. The design features are noted where appropriate, thus, a modification to that feature will affect the limiting characteristics and possibly the associated parameter peak value.

## Future Observation Type Helicopter Limiting Characteristics

Table XX presents the predicted occurrence of the parameter peak values for helicopters with limiting characteristics differing from those of the OH-6A. Several configuration changes thought to be likely for future observation type helicopters were selected. The modifications are: control system boost, quieter main and tail rotor blades, crew station vibration isolation and auxiliary lift surfaces. Following this, the OH-6A limiting characteristics and parameters affected by the change were determined. For example, the parameters of airspeed, load factor, and rpm are affected by the addition of control system boost because the primary OH-6A limiting characteristic of vibration in the cyclic stick due to retreating blade tip stall would not be present. Use of secondary or alternate limiting characteristics and their effect on high or peak value occurrence were also evaluated. Reduced recognition of certain parameter peak values would result from the addition of control system boost, quieter tail rotor blades and crew station vibration isolation. Therefore, new means of recognizing the impending peak values were considered for those configuration changes. For the configuration changes considered, no change in

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| TABLE XIX. OH-6A CHARACTERISTICS WHICH INFLUENCE MAGNI-TUDE OF PEAK VALUES OF LOAD PARAMETERS | |
|---|---|
| Parameter | Limiting Characteristics/Component(s) or Design Feature(s) Responsible |
| I. AIRSPEED | |
| A. Minimum | 1. Aircraft settling |
| | 2. Increasing pedal requirement |
| B. Maximum | 1. Pilot visual clues possibly including a high rate of descent |
| | 2. Retreating blade tip stall/rotor system design |
| | a. Noise |
| | b. Some pitchup |
| | c. Vibration in cyclic stick and possibly in collective stick and airframe |
| | 3. Structural/canopy design (could be encountered only at low density altitudes) |
| II. LOAD FACTOR | |
| A. Minimum | 1. Pilot physical clues |
| B. Maximum | 1. Pilot physical clues |
| | 2. Same as I.-B. 2 above |
| | 3. Bank angle in turns |
| III. LONGITUDINAL CYCLIC STICK | |
| Minimum and Maximum | 1. Droop stop contact (on ground only) |
| | 2. Excessive attitude change |
| IV. COLLECTIVE STICK | |
| A. Minimum | 1. rpm climbing (rotor uncouples from engine and begins to autorotate) |
| B. Maximum | 1. rpm drooping |
| V. ENGINE TORQUE PRESSURE | |
| Maximum | 1. High collective position |
| | 2. rpm bleed off/engine power available |

| TABLE XIX - Continued | | |
|---|---|---|
| Parameter | | Limiting Characteristics/Component(s) or Design Feature(s) Responsible |
| VI. RPM | | |
| A. Maximum | 1. | Tail rotor noise/tail rotor design |
| | 2. | Advancing blade tip Mach number (drag rise) in extreme cases only/rotor system design |
| | 3. | Transmission noise/transmission design |
| B. Minimum | 1. | High collective download (autorotation), occurs at high gross weights and airspeeds |

high or peak value occurrence is anticipated for the case where the alternate/secondary limiting characteristics do not affect the ability to recognize peak value occurrence. Similarly, for the case where a new limiting characteristic is recommended no change in high or peak value occurrence will result.

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TABLE XX. PREDICTION OF PEAK VALUES FOR HELICOPTERS DIFFERING FROM OH-6A

| Configuration Change | Limiting Characteristics Affected (Refer to Table XIX) | Parameters Affected | Alternate/Secondary Limiting Characteristics (Refer to Table XIX) | Remarks - Alternate/Secondary Limiting Characteristics Effect on Occurrence of Peak Values | New Limiting Characteristic | Remarks - New Limiting Characteristic Effect |
|---|---|---|---|---|---|---|
| 1. Control System Boost | I-B 2c*<br>II-B 2c*<br>VI-B 1<br>VI-B 2c* | Airspeed<br>Load Factor<br>rpm | I-B 1, 2a, b, 3<br>II-B 1, 2a, b, 3<br>VI-B 2a, b | Reduced recognition. Would increase frequency of high values and magnitude of peak values for airspeed and load factor. Would increase frequency and magnitude of low values of rpm. | "Cruise guide" indicator | No change in frequency or magnitude of peak values. |
| 2. Quieter Main Rotor Blades | I-B 2a<br>II-B 2a<br>VI-B 2a | Airspeed<br>Load Factor<br>rpm | I-B 1, 2b, c*, 3<br>II-B 1, 2b, c*, 3<br>VI-B 1, 2b, c* | Negligible change as primary limiting characteristic is still in effect. | None required | - |
| 3. Quieter Tail Rotor Blades | VI-A 1* | rpm | VI-A 2 | Reduced recognition of high rpm | High rpm warning system | No change in high or peak values. |
| 4. Crew Station Vibration Isolation | I-B 2c*<br>II-B 2c*<br>VI-B 2c* | Airspeed<br>Load Factor<br>rpm | I-B 1, 2a, b, 3<br>II-B 1, 2a, b, 3<br>VI-B 1, 2a, b | Same as (1) above. | Same as (1) above | Same as (1) above. |
| 5. Auxiliary Lift Surfaces | None | Airspeed | All limiting characteristics remain the same | Higher $V_{NE}$ would result from configuration change. No change in ability to recognize impending high air speeds. | None | - |

NOTE: Configuration changes considered separately.

*Primary or most effective limiting characteristics.

# TASK V - INDICATED REVISIONS TO DESIGN CRITERIA
## FOR OBSERVATION TYPE HELICOPTERS

### INTRODUCTION

This task consisted of reviewing the results of Tasks I through IV, and determining the indicated revisions, if any, to be recommended in regard to the design criteria for observation type helicopters. The recommendations resulting from the study are noted in the following paragraphs, categorized by task number, plus some of more general nature.

### RESULTS OF THE STUDY

#### Task I

The design mission profile was found to most nearly match the operational flight spectrum as developed from TR 71-60. The AR-56 spectrum showed poorer agreement, and the AMCP 706-203 spectrum the poorest agreement. Therefore, the operational spectrum derived in Task I is recommended as the one to use in the design criteria for future observation type helicopters.

#### Task II

The comparison of fatigue load spectra, fatigue damage rates and fatigue lives between the predicted and the operational mission profiles indicates the following:

a.  Main Rotor Blade - Relatively excellent agreement was found in regard to the affected parameters between the predicted and the operational mission profiles as far as the main rotor blades are concerned.

b.  Tail Rotor - During the OH-6A engineering development flight strain survey, it was found that the only fatigue damaging flight conditions for the tail rotor were extreme yaw maneuvers and ground-air-ground cycles. Unfortunately, sufficient data were not obtained in these areas during the operational loads program (TR 71-60) to provide any reliable comparison of the stated parameters as far as the tail rotor is concerned.

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Therefore, the design mission profile was satisfactory in determining the service lives for the OH-6A Helicopter, and the operational mission profile is recommended as suitable for the design criteria for future observation type helicopters. AMCP 706-203 should be revised and updated.

Task III

Analysis of historical changes in fatigue lives and configurations indicated the following:

a. Main and Tail Rotors - No indication exists of any inadequacy of the design fatigue load spectrum with respect to the main rotor blades or the tail rotor blades.

b. Rotor Drive System - Considerable historical indication was found to exist that both the main and tail rotor drive systems encountered a more severe load (torque) spectrum in service than the predicted spectrum based on the design torque spectrum and the engineering development flight strain survey loads. This historical indication was only partially confirmed by the data in TR 71-60, possibly due to inadequate data provided on sideward and yawed flight conditions, which are critical for the tail rotor drive train. The data in TR 71-60 does indicate that a significant portion of flight time was spent at higher gross weights and higher power settings than anticipated by the original design specification. There is also the possibility that some ships in the operating fleet were operated at higher powers and gross weights than were the instrumented ships involved in the operational loads program of TR 71-60.

Therefore, an increase in severity of the torque spectrum over the OH-6A design spectrum for both the main and tail rotor drive systems is recommended as a result of the review of historical data. No such historical indication was found in regard to the main rotor blade or tail rotor load spectrums.

Task IV

Based on the good correlation of the operational peak value data arrived at under Task IV with the design peak values (i.e., design limit values) for the OH-6A, no changes in the design criteria for peak (limit) values are recommended.

## ADDITIONAL COMMENTS AND RECOMMENDATIONS

With regard to main rotor blade fatigue damage, the AR-56 and AMCP 706-203 flight load spectrums resulted in no damage for some conditions that were found to be damaging for both the design spectrum and the operational spectrum. Also, some damaging flight conditions in both the design and operational spectrums of the OH-6A were not included in the AR-56 or AMCP 706-203 spectrums. This observation explains at least partly why these latter spectrums resulted in less conservative predictions of the safe fatigue service life.

The recording and presentation (in TR 71-60) of the various flight spectrum parameters independently, rather than simultaneously, resulted in reduced ability to reliably deduce the relative occurrence of adverse versus favorable combinations of the recorded parameters. For example, the degree of blade stall in severe maneuvers (and thus, the severity of the blade fatigue loads) often depends upon a peculiar combination of several of the measured parameters. Prediction of occurrences of the more severe blade load producing combinations of parameters, therefore, involved some ordering of the data for the independent parameters, including simplifying assumptions and engineering judgement, which tended to reduce the reliability of the analysis. Hence, it is recommended that, in future operational flight loads programs, more effort be made to record and reduce the obtained data in a form as close as possible to actual load spectrum data for the major components or systems.

More specific data should be recorded in regard to ground-air-ground cycles. Counts of both complete stop/starts, additional cycles down to engine idle, needle splits and touchdown/liftoffs should be presented since, for the rotating systems of the helicopter, this often is one of the few damaging fatigue conditions.

More specific data on cumulative time at torque levels should be obtained for all the major branches of the main rotor and tail rotor drive system.

Load spectrum and damage rate analysis (under Task II) indicates that mathematically simplified approximations of the fatigue load spectrum and fatigue damage rates may be useful for both predictions and comparisons.

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